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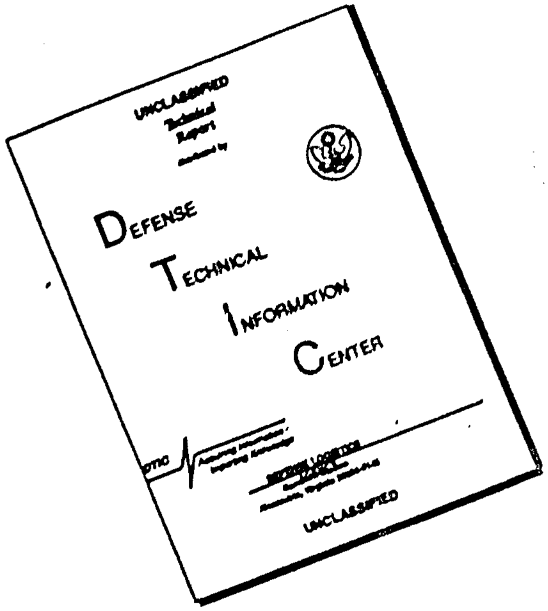

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AIR FORCE FLIGHT TEST CENTER
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UNITED STATES AIR FORCE
EDWARDS AIR FORCE BASE, CALIFORNIA

AD INEXPLORATA

AD 256791

REPLY TO
ATTN OF: FTAT/Mr. Lemmon/46181

SUBJECT: Publishing Error

23 August 1961

TO: ASTIA (TIPCR)
Arlington Hall Sta
Arlington 12, Va

1. It has been noted that a publishing error exists in AFFTC-TR-61-1, "YHC-1A Flight Evaluation."

2. The numbering and location of pages 27 and 28 are incorrect. Page 27 should be page 28 and page 28 should be 27.

3. It is requested that this letter be made a part of the AFFTC-TR-61-1 documents in your file.

FOR THE COMMANDER

J. A. Statz

J. A. STATZ
Major, USAF
Director of Administrative Services

FOR ERRATA

AD 256 791

THE FOLLOWING PAGES ARE CHANGES

TO BASIC DOCUMENT

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AFFTC-TR-61-1
February 1961

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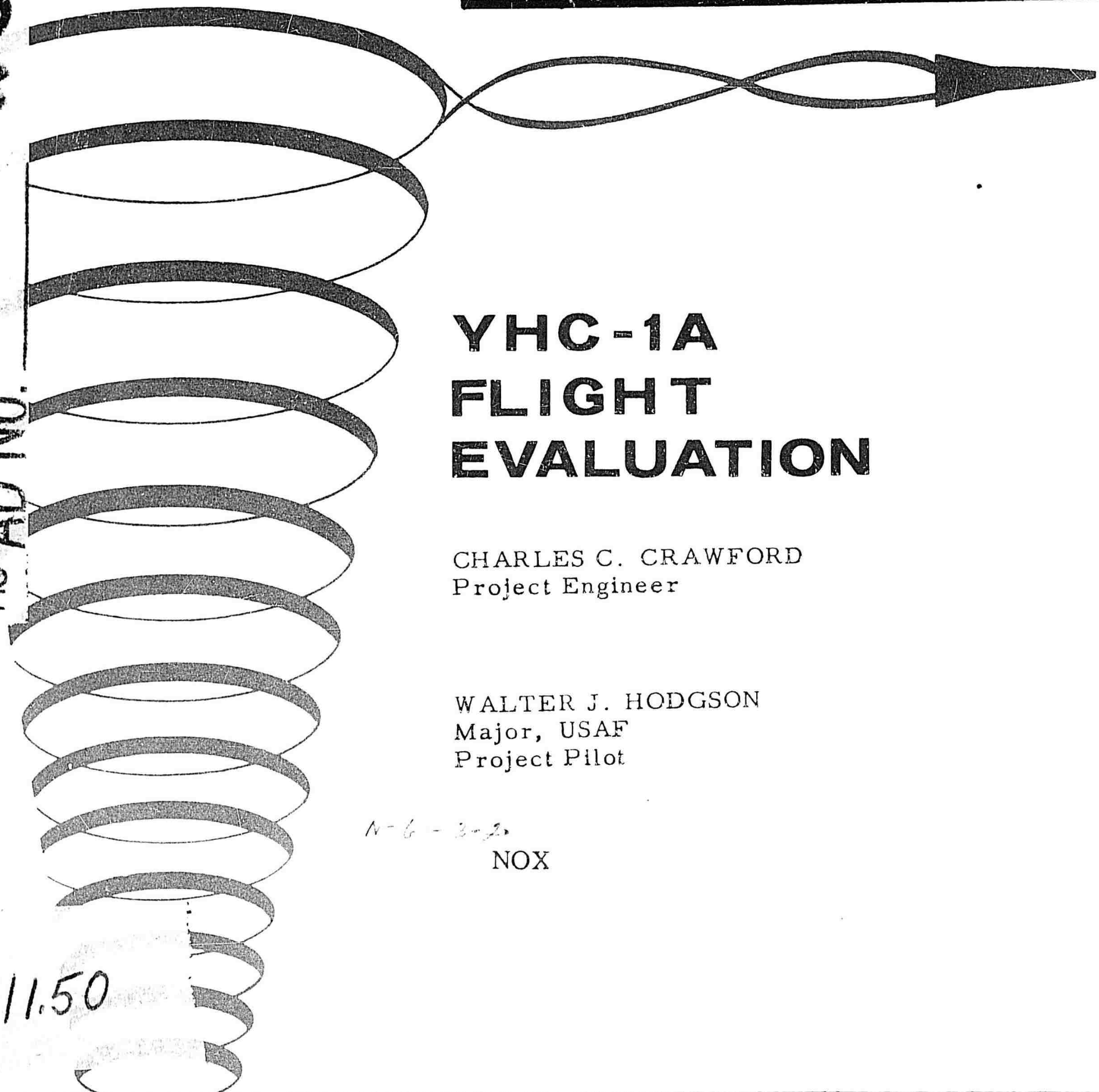

YHC-1A FLIGHT EVALUATION

CHARLES C. CRAWFORD
Project Engineer

WALTER J. HODGSON
Major, USAF
Project Pilot

N-6-3-2
NOX

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

AFFTC-TR-61-1
February 1961

YHC-1A FLIGHT EVALUATION

CHARLES C. CRAWFORD
Project Engineer

WALTER J. HODGSON
Major, USAF
Project Pilot

This report has been reviewed and approved

Clayton L. Peterson
CLAYTON L. PETERSON
Colonel, USAF
Director, Flight Test

[illegible], Col, Dep, Cmd

JOHN W. CARPENTER, III
Brigadier General, USAF
Commander

ABSTRACT

This report presents the results of the Air Force Flight Test Center Flight Evaluation of the YHC-1A helicopter. Twenty-five flights for a total of 22 hours and 35 minutes were flown between 28 June and 10 September 1960 to evaluate the general flying qualities of this vehicle. Particular emphasis was placed on evaluation of the stability augmentation system (SAS). Very limited performance data was obtained.

The YHC-1A is the second model of the Vertol 107 series and is a twin turbine, tandem rotor, tactical transport helicopter manufactured by the Vertol Division of the Boeing Airplance Company. It is powered by two General Electric T58-GE-6 free turbine engines and has a design gross weight of 15, 550 pounds.

The YHC-1A represents an advance in the art of helicopter design due to its excellent handling characteristics, positive dynamic stability, and low vibration levels at high speed. The rear ramp loading facilities, two-engine reliability, excellent maintenance accessibility and high ratio of cargo volume to airframe volume with adequate weight lifting capabilities makes the YHC-1A potentially the most operationally suited of any modern helicopter tested by the AFFTC.

Several desirable features of this helicopter include the near level fuselage attitude at cruise speed, excellent cockpit emergency exits, general cockpit layout, and the automatic response of the second engine and lack of aircraft trim disturbance following an engine failure.

The major undesirable features of the YHC-1A include its dependence on the SAS for adequate stability and control, inability to utilize full engine power due to transmission torque limits, excessive rpm droop during power application which can lead to the loss of generator output and a dual SAS failure, slow beep trim rate, the location and operation of the rotor brake and parking brake, the lack of a manual fuel control shutoff, lack of pilot control of the fuel boost pumps, and lack of engine anti-icing and fire extinguishing equipment.

Additional shortcomings include poor static directional stability during partial power descents and autorotation.

Although the probability of a complete SAS failure is remote, a SAS off flight was made. SAS off operation of the YHC-1A is possible; however, stability characteristics are marginal for flight in moderate turbulence and unacceptable for flight under instrument conditions in that the possibility of safely returning to the base is considered remote.

It should be pointed out that many of the problem areas found to exist in the YHC-1A may be eliminated by already programmed design changes for the Vertol 107 Model II which is now in production.

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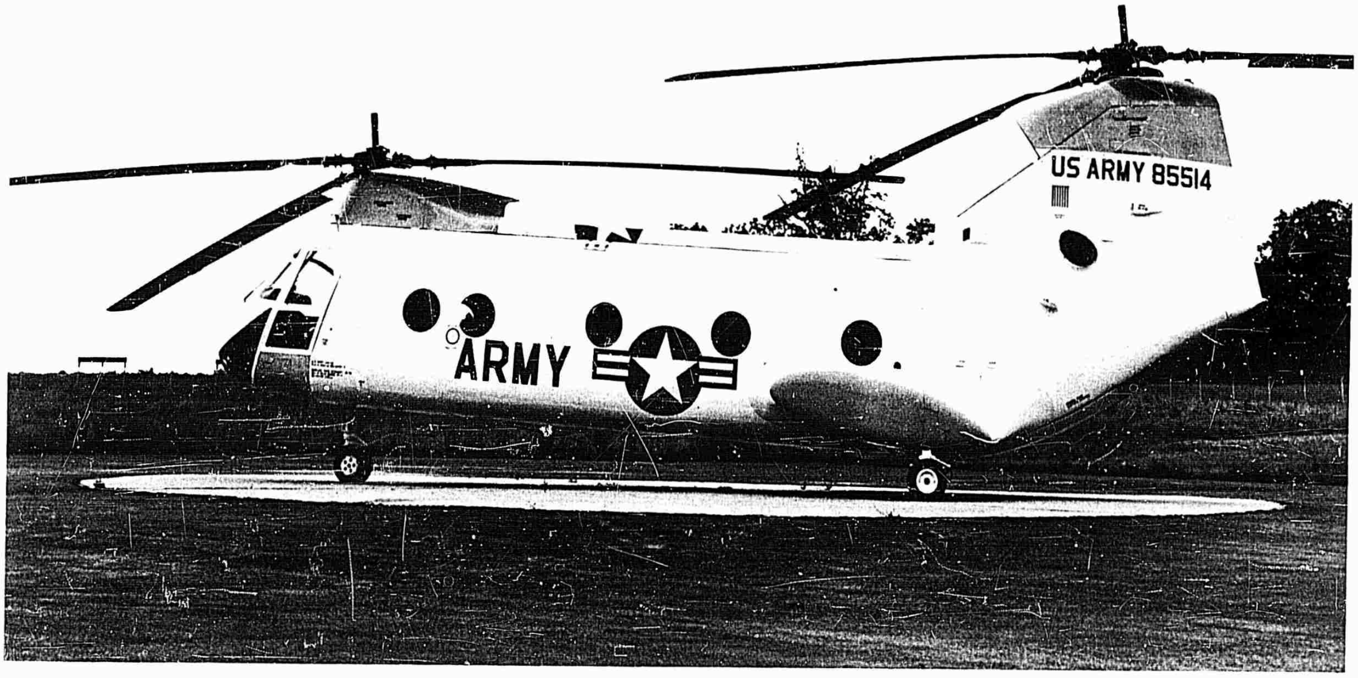
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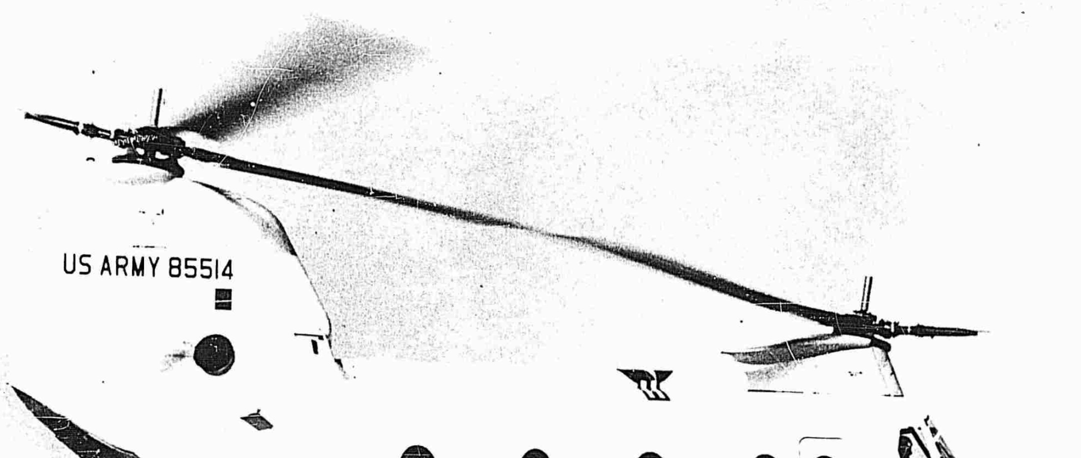
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INTRODUCTION

This report presents the results of the Air Force Flight Test Center Flight Evaluation of the YHC-1A helicopter. This evaluation was conducted at the contractor's Philadelphia facility. Twenty five flights for a total of 22 hours and 35 minutes were flown on aircraft S/N 58-5514 between 28 June and 7 July and between 4 and 10 September 1960.

The purpose of this flight evaluation was to evaluate the general flying qualities of the YHC-1A. Particular emphasis was placed on evaluation of the stability augmentation system (SAS) because of its simularity to the system being developed for the YHC-1B. Very limited performance data was also obtained.

The YHC-1A is the second model of the Vertol 107 series and is a twin turbine, tandem rotor (two 3-bladed rotors), tactical transport helicopter manufactured by the Vertol Division of the Boeing Airplane Company. Three YHC-1A helicopters were built for the U.S. Army and feature rear ramp loading and the capability of carrying a crew of three with 26 passengers or a crew of three with 15 litter patients and two medical attendants.

This helicopter is powered by two General Electric T58-GE-6 free turbine engines rated at 1050 SHP at sea level. A common gear box transmits the power to an interconnect shaft which drives the forward and aft transmission and rotor. The transmissions are limited by stress to a torque equivalent to 1700 SHP at 258 rotor rpm. All electrical power is supplied by two 20 KVA alternators. Direct current requirements are fed by transformer rectifiers which in turn are fed by the alternators. Hydraulic pressure in the aircraft is supplied by two hydraulic pumps delivering four gpm each at 1500 psi. Engine and transmission cooling is accomplished by an oil cooling system.

All YHC-1A primary flight controls are equipped with completely irreversible hydraulic boost. To ensure reliability, the boost system is duplicated. The primary or upper system derives its power from the rotor side of the front transmission and the secondary or lower system from a separate pump also mounted on the rotor side of the front transmission. Both systems are operating under normal conditions.

Artificial feel is provided to the cyclic control stick through springs attached to the stick and to magnetic brakes. Force trimming is accomplished by depressing a button on the cyclic grip which releases the magnetic brakes and allows the springs to re-center.

Longitudinal control is provided by means of collective pitch which is applied to the front and rear rotor in a differential manner. Rearward longitudinal movement of the stick results in an increase of collective pitch on the front rotor and an equal decrease in collective pitch on the rear rotor.

Longitudinal cyclic pitch of the front rotor is fixed at 2 degrees forward. Rear rotor longitudinal cyclic pitch is automatically varied, as a function of dynamic pressure, from 0.5 degrees (aft) at hover to 7.5 degrees (fwd) at 140 knots IAS. Longitudinal cyclic pitch variation is used primarily to control rotor flapping and to provide a stable stick position - airspeed gradient.

Lateral control is provided by means of lateral cyclic pitch which is applied to both rotors simultaneously. Right lateral movement of the stick results in the application of right cyclic pitch to both rotors. The cyclic pitch application is unsymmetrical; more lateral cyclic pitch is applied to the front rotor than the aft rotor to ensure good pedal-fixed turn capabilities.

Directional control is provided by means of lateral cyclic pitch which is applied to both rotors in a differential manner. Forward movement of the right pedal results in the introduction of right lateral cyclic pitch in the front rotor and an equal amount of left lateral cyclic pitch in the rear rotor.

The stability augmentation system (SAS) provided on the YHC-1A is an electro-hydromechanical servomechanism which augments the stability of the basic helicopter.

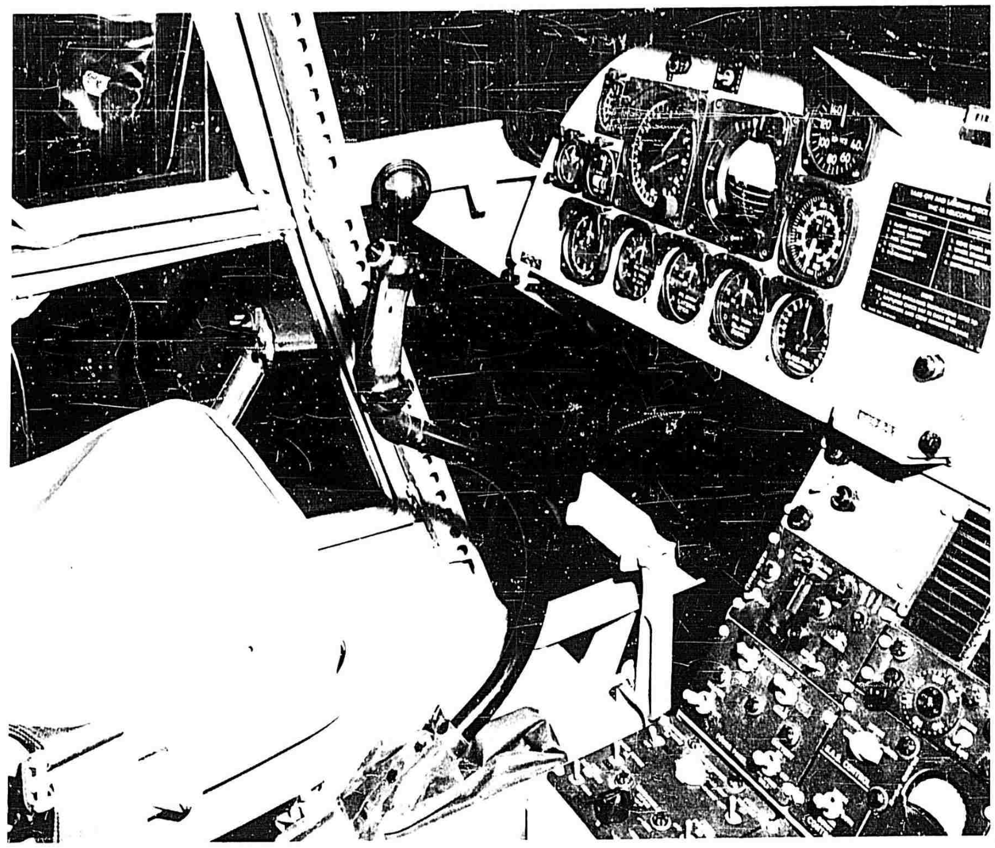
SAS control inputs are made through differential actuators and do not move or otherwise disturb the cockpit controls. Whenever a SAS control input is made, the control stops in the cockpit are automatically shifted an amount equal to the SAS input, thus giving the pilot the capability of completely overriding the SAS if he so desires.

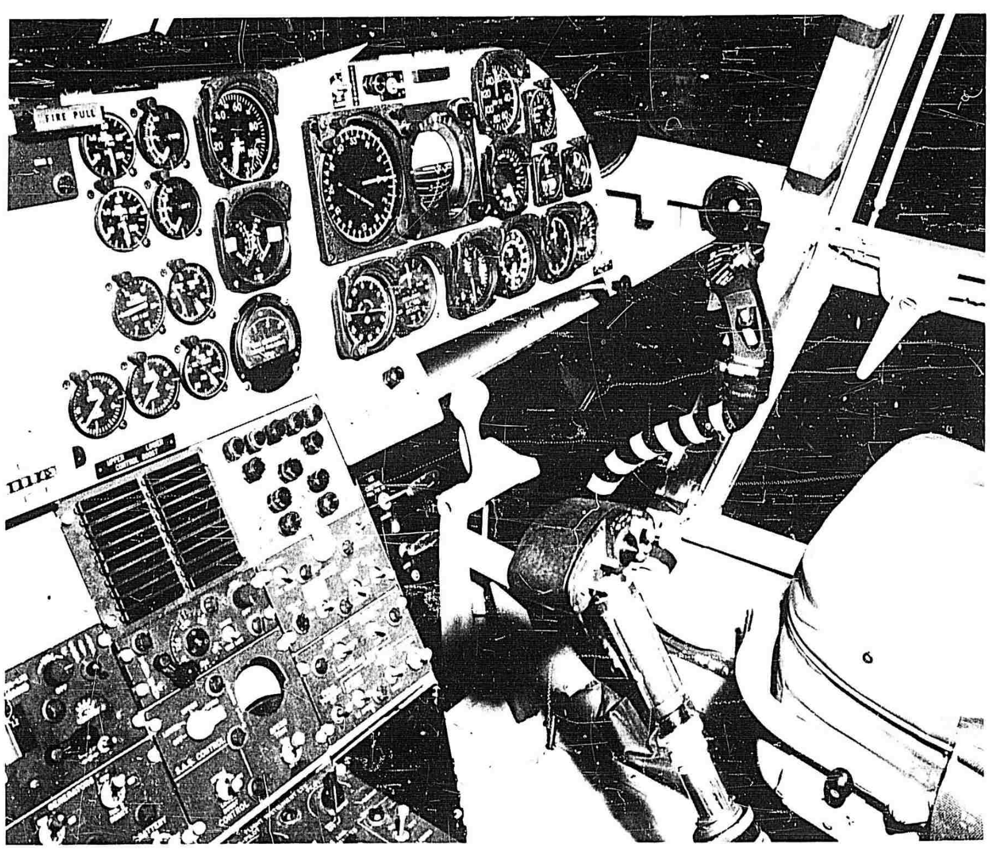
The SAS system senses angular velocity about all axes and sideslip angle. A detailed description of the flight control system is presented in Appendix II.

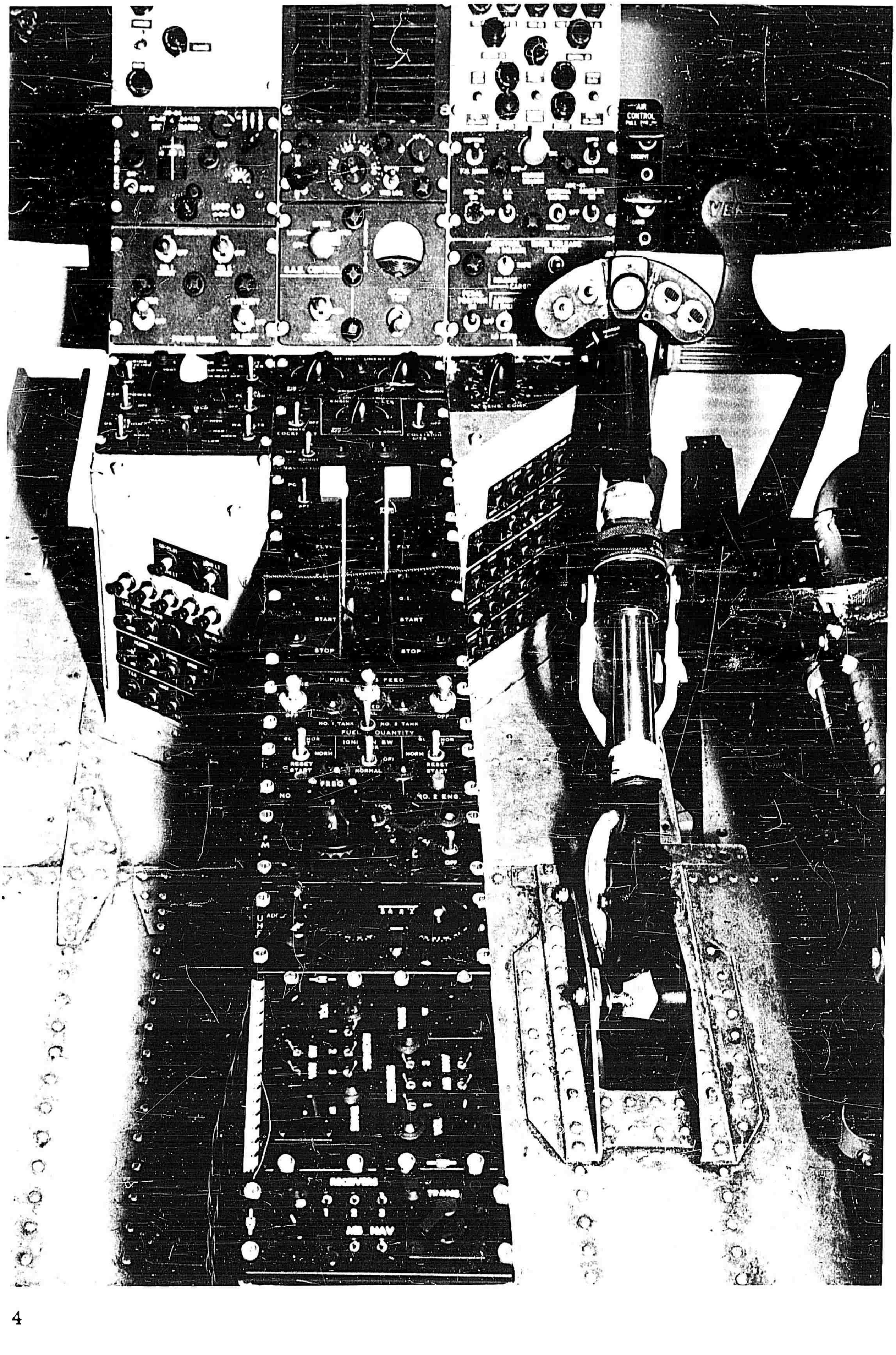
The basic weight of a production aircraft is 8946 pounds with a fuel capacity of 345 gallons (2240 pounds). The basic weight of the test aircraft was 12,530 pounds because of over 3100 pounds of test instrumentation. The design operating gross weight of the helicopter is 15,550 pounds. At this gross weight, the premissible c g travel is from 24 inches forward to 5.5 inches aft of the centerline between the rotors. This helicopter was tested from the forward limit to 2 inches aft.

The only external changes to the helicopter for test purposes consisted of the addition of a large nose boom containing a sideslip vane and external housings for the torquemeter above each rotor shaft. Both of the items have a small drag penality; however, they should not affect stability and control.

The data obtained during these tests was given the contractor as it was obtained. A set of final plots as presented in the report was sent to the contractor in January 1961.




FIRE PULL


AIR CONTROL
FLY
START
STOP
FUEL FEED
NO. 1 TANK
NO. 2 TANK
FUEL QUANTITY
RESET START
NORMAL
NO. 2 ENG.
UHF
RECEIVERS

TEST RESULTS

COCKPIT EVALUATION

In general, the cockpit facilities and equipment of the YHC-1A are satisfactory.

Entrance to the cockpit is provided only from the cabin section. Normal entrance to the helicopter cabin is provided through the loading ramp. Entrance to the cockpit through the cabin section will compromise the capability of loading bulky cargo or restrict placement of cargo. Direct entrance to at least the pilot's seat should be provided from outside the helicopter. Actuation and proper sequencing of the electrically actuated ramp doors is provided both inside and outside the helicopter. The electrical power source to operate these doors is available only from inside the helicopter. Provision should be made to operate the exterior opening without requiring auxiliary power or entrance to the helicopter. A door located at the right forward section of the fuselage may be used for an additional entrance; however, entrance is awkward since the door level is approximately 4 feet above ground level and steps or hand holds are not provided. The door pulls inward and slides on tracks to the top of the cabin roof. Operation of the door should be improved since it is difficult to open from the outside and it is possible to jam the door in the full open position or slide it out of its tracks during closing, thereby preventing locking of the door. (B-23, C-12, C-4, and B-25)[1]

The cockpit entrance is well padded, free of obstructions, and of sufficient size to allow ease of entrance or exit while wearing a parachute. Entrance to the pilot's seat is awkward due to the long center console and position of the collective pitch control. Installation of a flat plate between the console and the collective pitch attachment bracket would provide a convenient step to aid entrance. The pilots' seats are comfortable and are adjustable fore and aft and vertically. (C-14)

Collective pitch position, range of travel, and lack of physical restrictions are excellent. The collective pitch control is not balanced, and friction must be applied to keep the collective pitch from creeping down during flight. The unbalance of force required may lead to overcontrolling with collective pitch during small movements required during a hover. A throttle twist grip is not provided. Collective pitch friction operates satisfactorily and small adjustments are possible without binding.

The control head on the end of the collective pitch control contains switches at a constant radius from the hand grip and contains two power turbine selector (beep trim) switches, landing light switches, a search

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[1] Numbers indicated as (B-23, etc.) represent the corresponding recommendation numbers as tabulated in the recommendation section of this report.

light switch and a button to convert the function of the search light switch for operation of a retractable mirror. Operation of all switches was satisfactory with the exception of the mirror button which required too much force for actuation. (C-13)

The left beep trim switch actuates both engines simultaneously, while the right switch controls only the number two engine to match the power output or speed of the number one engine. Location of the switches is adequate for ease of operation.

The cyclic control height is comfortable and displacements, while not in accordance with HIAD, were adequate for all flight conditions encountered. Adequate clearance is available between the cyclic control and the collective pitch control under the most critical conditions.

The operation of the side windows is unsatisfactory. The windows are hinged at the top, open outward, and are held open with two braces secured by friction screws. The windows can not be left open for ventilation during forward flight. Operation of the window requires two hands and locking is extremely difficult. (B-24)

An emergency exit is provided by large panels adjacent to each pilot's seat. Operation of the release is quite easy, and the size and accessibility of the opening is excellent. The emergency release handle is located on the jettisonable door. Location of this handle on the fuselage is desirable. In the cabin section, emergency exit kickout panels are provided in the upper ramp door and the forward door.

A crew member's seat is provided which folds down across the cockpit entrance. The seat does not incorporate a back rest or guard which would preclude any possibility of the crew member sliding aft out of the seat belt. The headroom is inadequate when wearing a protective helmet and occupying this seat becomes extremely tiring after a short period of time. (B-16)

The fuel and engine control panel is on the center console and is conveniently located adjacent to the collective pitch stick. This panel consists of a fuel quantity test switch, fuel switches for each tank, a crossfeed switch, starter buttons, governor overspeed test switches and two condition levers. The condition levers are spring loaded to the outboard edge of each slot with detent positions labeled "STOP", "START", "GROUND IDLE", "FLIGHT IDLE" and "FLY". The detents provided for the intermediate lever positions are not adequate for proper control of the levers. It is possible to advance the condition lever from the "START" position to the "FLY" position without contacting the intermediate detents. When retarding the condition lever, the "GROUND IDLE" detent is not adequate to prevent the lever from being retarded to the "STOP" position. In addition, the condition lever was scoring at the point of contact with the edge of the guide slot. (B-7)

Adequate stops should be provided for all lever postions while advancing the condition lever, and a positive stop provided at the "GROUND IDLE" position while retarding the lever (Reference HIAD J, 2-2.6-3).

The center console should be rearranged as follows:

1. Relocate the master, battery, and generator switch panel to center of the console for accessability. (B-19)

2. Relocate the light control panel to the left of the console. (C-5)

3. Re-label "SAS CONTROL" and "BOOST CONTROL" functional arrangements to prevent confusion. At present, the "SAS CONTROL" label is too close to the boost control, and the "BOOST" label is in smaller letters than the "CONTOL" label. These labels should also be located above the appropriate control. (B-20)

4. Provide adequate identification of circuit breakers without restricting visibility of the circuit breaker. (C-6)

5. Provide a warning light dim switch in accordance with HIAD. Present system dims when pilot's flight instrument light switch is "ON". This should be a separate circuit, returning to bright upon interruption of electrical power. (C-7)

6. Reverse the ignition switch actuation to have the switch forward while in the "Normal" position in accordance with MIL-STD-250A, 24 July 1956, Paragraph 1.3.a.(1). (C-8)

7. Provide lighting for the circuit breaker control panel. (C-9)

8. Provide an "APU CONNECTED" word warning signal. (C-1)

9. Provide an "ENGINE LOW OIL PRESSURE" word warning signal. (B-6)

10. Provide a "SINGLE SAS" word warning signal. (A-5)

11. Provide ramp actuation control switches in the cockpit. (B-21)

12. The UHF and VHF radios require manual frequency selection. Holders for channel frequency cards adjacent to these sets are necessary. (C-10)

The instrument panel is generally acceptable; however, the incorporation of the military standard flight instrument arrangement for helicopters (MS 33572, dated 13 November 1959) would improve the presentation. The master warning light is too dim for ready visibility. (C-2, B-8)

Engine torquemeters are not provided due to lack of reliability; however, due to transmission torque limitations, torquemeters for each transmission, indicating from 60 percent to 110 percent, are provided and operate adequately. Variance of transmission torque requirements from hover to forward flight required incorporation of these torquemeters which allow maximum use of engine power through transmission torque distribution. An automatic load sharing device should be incorporated for the engines. (B-4)

Engine torquemeters should be incorporated for determination of engine performance. Fuel flowmeters should be incorporated for matching engine fuel consumption and rapid determination of range capabilities. (B-15)

The emergency fluid shut-off "T" handles illuminate and operate satisfactorily; however, they are too close together and may restrict single actuation or result in actuation of both levers which will result in dual engine failure. These handles should be moved further apart. (A-6)

Location and operation of the parking brake is unsatisfactory In accordance with C. 2-2.4.1.3. HIAD and Paragraph 1.2.b. and 2.14 MIL-STD-250A, dated 24 July 1956, "Location of the parking brake lever should be to the left of the pilot and located under the instrument panel. Operation of the parking brake will result from pulling the handle and release by depressing the foot brakes." The present configuration requires releasing the cyclic control and actuating a small lever located on the floor to the right of the pilot for both locking and releasing the parking brake. In addition, the warning light is activated by the lever position which can allow the light to be illuminated while the parking brake is unlocked or the light may be out when the brake is still locked. The addition of foot brakes in the copilot's station is required by C.2-2.4.1.3. HIAD. (A-10, C-3)

Location and operation of the rotor brake is unsatisfactory. Paragraph J. 2-2.4.3. HIAD and Paragraph 2.16 MIL-STD-25A, dated 24 July 1956, requires the brake to be located adjacent to the power quadrant available to both pilot and copilot and operable by the pilot's left hand with pull to actuate and push to release. The present installation is located against the aft bulkhead to the right of the pilot. Force of actuation is excessive, and positive locking is not assured unless excessive pressure is employed. Actuation of the rotor brake requires placing the lever in the full off position. On numerous occasions the pilot's headset cord prevented returning the lever to the full off position and braking action could not be obtained. The lever position of the rotor brake will actuate the warning light on the enunciator panel; however, rotor brake action is not assured except by manually checking the brake pressure by operating the handle. (B-9)

Neither engine anti-icing nor fire extinguishing capability were provided in the engines tested. Both are mandatory for service use. (A-7)

An emergency fuel control feature is not incorporated due to the dual engine installation. All engine and fuel control functions are actuated electrically by the controls in the cockpit. In the event of complete electrical failure or malfunction of the controls it will be impossible to shut down the engines from the cockpit. Incorporation of a manual fuel shut-off available to the pilot and copilot is desirable. Incorporation of an auxiliary power supply to operate the fuel controls is required to prevent compromise of engine control during operation below generator cut-out speed in the event the battery is dead. This system should not operate unless insufficient electical power is available to operate the fuel controls when the battery switch is on. It is desirable that control of the fuel boost pumps be available to the pilot. (A-8, B-29, and A-9)

The fuel crossfeed system is unsatisfactory since the boost pump with the most pressure will provide the crossfeed. This permits crossfeed from the low level fuel tank. Fuel boost pump switches are not provided and circuit breakers for the fuel pumps are located at the aft of the cabin. In order not to compromise cargo positioning in the cabin, all circuit breakers should be relocated to the cockpit. (A-14)

The directional pedals adjust fore and aft simultaneously. These adjustments generally allow selection of an excellent relationship of all flight controls for various size pilots or for pilot preference. The directional pedal adjustment is electrically actuated by a switch on the console and is difficult to locate. There is no individual adjustment of the pedals; however, the small directional pedal trim changes encountered throughout the flight regime do not require this feature. Several times during the program the pedal adjustment mechanism failed during toe brake application which resulted in compromise of braking action due to the extension of the pedals. The reliability of the locking mechanism must be improved to prevent compromise of aircraft control during ground handling, takeoff or landing. To further improve reliability and reduce complexity a mechanical adjustment should be substituted for the electrical adjustment feature. (B-22)

A small fan in the center of the cockpit ceiling is provided for ventilation and is inadequate. Use of the heater vent system provides more satisfactory ventilation. Additional ventilation would be provided if the windows could be left open during forward flight. (B-24)

The cockpit noise level is quite high and results primarily from the forward transmission located above and immediately aft of the cockpit. Communication is adequate with the interphone and radio system. Use of the cockpit heater vent fan further increases the noise level. It should be pointed out that the test vehicle was not equipped with production sound proofing.

Visibility is generally good; however, if the seat is lowered, visibility over the instrument panel becomes limited. For small pilots, additional seat height adjustment would be desirable. Raising the entire seat adjustment range is recommended. (C-15)

GROUND HANDLING

Taxi characteristics with the nose wheel on the ground are unsatisfactory. The forward tilt of both rotor shafts requires continual use of brakes to prevent or control forward motion. Use of aft cyclic control will merely result in a differential collective pitch change since longitudinal control by cyclic pitch change is not incorporated in this aircraft. The dual nose wheels do not rotate independently, which results in extreme difficulty in obtaining even a large turning radius with use of both directional pedal and differential braking. When utilizing directional or lateral control, it is also necessary to increase collective pitch in proportion to the control deflection to prevent striking the rotor blade droop stops. (B-26, A-12)

Taxiing by raising the nose wheel off the ground provides excellent control of the aircraft. The aircraft is quite stable in the nose high attitude and may be easily trimmed for hands-off operation. Turns are possible over a spot and taxiing both forward and aft is possible with the nose wheel elevated. A combination of aft longitudinal control and braking is necessary for effectively stopping the aircraft when the nose wheel is off the ground. The rate of deceleration with aft cyclic, without an excessively nose high attitude, is too slow and brake pressure required in relation to brake effectiveness is high.

During rotor engagement, ground runup and shutdown with winds approximately 10 knots or above, it is necessary to use a slight amount of collective pitch to prevent the rotor blades from striking the droop stops. Positioning of the controls in other than neutral position will further aggravate the interference.

The cabin or cargo section size, lack of protuberances of essential equipment (with the exception of the circuit panels), and the rear loading ramp, which can remain open during flight, permits the use of this helicopter for a wide variety of missions.

A winch located in the forward cabin bulkhead may be used by a system of pulleys for hoisting cargo through the ramp doors or hoisting through an opening in the center of the fuselage.

The lower doors in the fuselage opening should extend about 45 degrees beyond the vertical to prevent damage of the doors if the hoist or sling cable should strike them. (B-18)

Interphone connections are available at the forward, center, and aft sections of the cabin area. A communications cord should be provided with sufficient length for a crew member to be stationed outside the helicopter for fireguard duty during starting and to enter the cockpit without changing headsets. (C-11)

ENGINE START AND RUN-UP

Engine start is relatively simple. A hold-down relay is provided for the start switch. For battery starts the battery switch is placed in the "D.C. START" position which activates a small inverter to operate the necessary a.c. instruments. This inverter operates automatically when auxiliary power is utilized and is necessary since a.c. current is normally supplied by the transmission driven generators.

Neither ignition nor fuel is supplied to the engine when the condition lever is in the full off position. Placing the condition lever in the start position arms the starter hold-down relay and supplies ignition when the starter button is depressed.

The engine will accelerate to approximately 20 percent rpm on starter alone. Normally, at 15 percent engine speed the condition lever is advanced to the ground idle position which allows fuel to be supplied to the engine. Following combustion the acceleration to flight idle speed of 40 to 60 percent is automatic. The starter will cut-out automatically at approximately 40 percent. The critical phase of the starting sequence is at approximately 40 percent engine speed, and is characterized by a marked decrease in engine acceleration. If fuel scheduling is too rich, initial acceleration too slow, or initial exhaust gas temperature above 100 degrees C an overtemperature condition may result and require terminating the start. Stopping the start at any time is accomplished by retarding the condition lever to the "STOP" position. If fuel scheduling is too lean, the engine will not accelerate beyond approximately 40 percent, but an overtemperature condition will not result. In this event a fuel control adjustment may be necessary since an additional method of adding throttle is not provided. A normal start is completed in approximately 25 seconds.

Following engine start, the rotors are engaged by merely releasing the rotor brake. Acceleration to ground idle speed of approximately 105 rotor rpm is satisfactory. The front rotor blade may be a ground clearance problem during initial acceleration or shutdown. Clearance, with blade droop, is approximately 7 feet and may become less during gusty wind conditions. The blades will cone up sufficiently to provide adequate ground clearance at ground idle speeds.

Advancing the condition levers from ground idle to flight idle will result in a rotor acceleration that is excessive. The sudden increase in noise and vibration is disconcerting. The rotor speed will increase from 105 rpm to 240 rpm in approximately 6 seconds; however, an overtorque condition does not result. Since the condition lever is essentially an electrical switch, this acceleration cannot be controlled. Accelerating the rotors by advancing only one engine does not appreciably decrease the time required. (B-10)

The flight idle position results in an engine speed of approximately 80 percent and rotor speed at the minimum available in the normal operating range (240 rpm). In this position, the governor speed selector (beep) switches do not operate. This speed also corresponds closely to the minimum speed at which the a.c. generators will operate. If the beep switches are in their full low actuation range, there will be no change in rotor rpm as the condition lever is advanced to the "FLY" position. If the beep switches are in other than the low setting, rotor rpm will accelerate to the speed corresponding to the governed setting.

The flight idle position of the condition levers does not provide engine/rotor governing and is not a necessary position. The flight idle position may be a hazard to flight since it provides a flight range rotor rpm and may be inadvertently left in this position during take-off. This will result in low rotor rpm, loss of the a.c. generators, and failure of both stability augmentation systems. (B-17)

The stability augmentation systems should not normally be turned on until the remote indicating artificial horizons are up to speed and indicating proper operation. This system results in delays on take-off, since the gyros for the artificial horizon also operate the SAS and will not begin operation until the a.c. generators are on the line. The gyros for the artificial horizons will normally be operational in approximately 2 minutes following connection of the a.c. generators; however, in the event the gyro is still coasting down from a previous operation or has tumbled, the normal fast erection feature will not function and approximately 15 to 20 minutes may be required for the artificial horizon to erect itself. Although instrument capability is compromised, the SAS may be turned on when the artificial horizon is indicating operating speed (noted by the disappearance of an "OFF" indicator), since the slow rate of erection by the gyro will not be sensed by the SAS. The deficiencies in the attitude sensing system should be corrected. In addition, separate rate gyros to operate the SAS components are necessary to prevent compromise of SAS operation due to discrepancies in the attitude sensing system. (A-11)


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TAKE-OFF, HOVER, TRANSITION AND CLIMB

To prevent rolling forward during collective pitch application, it is necessary to maintain braking action until the nose-up attitude is sufficient to place the plane of the rotors parallel to the ground. Aircraft stability during lift-off is very good. There is negligible differential oleo strut action to disturb lateral control, and change of directional control is not necessary with addition of power. Approximately one inch aft cyclic is required to hover with a mid cg. Cyclic control position is comfortable. Lateral and longitudinal control harmony and sensitivity are good. Directional control sensitivity is slightly low; however, adequate control is available. The sensitivity of the collective pitch is high, and since the collective system is not balanced, friction is required to prevent downward movement. With collective friction applied, breakout forces combined with high sensitivity may lead to slight overcontrolling in a hover.

Stability and control during normal hovering is very good. During precise hovering, such as sling load pickup or spot landings, continual small rapid lateral and longitudinal movements were noted, even in calm air. Correction of these movements was effective and on numerous occasions the helicopter would return to the position it was in prior to the correction. During these small movements, pitching or rolling of the aircraft was not noticeable. These movements may be due to fuselage reaction to rotor downwash, SAS corrections for minor disturbances, or false signals introduced to the SAS system. The cause of these movements should be determined and eliminated. (B-27)

The cyclic longitudinal breakout forces are slightly high for the sensitivity provided. This results in overcontrolling during precision hovering unless the trim release button is depressed. A trim release switch should be provided to eliminate the cyclic trim force system as desired. The trim force gradient also prevents trimming of small residual forces. (B-28)

Engine droop characteristics vary considerably with rpm and power required (collective pitch position). With collective pitch full down, the governing range is from 94 percent to 108 percent (242-278 rpm). During heavy weight hover, governing rpm range is from 97 percent to 104½ percent (250-269 rpm). A loss of 10 rpm is encountered at full beep when collective pitch is increased from full low to that required for hover. At full low governed rpm, a gain of approximately 6 rpm is encountered from full down collective to the hover position. Throughout the test program, the droop characteristics were not consistent. The droop eliminating system, which has electrical sensing, should be capable of maintaining a maximum droop of 1 percent rotor speed (2½ rotor rpm) throughout the allowable governor rpm range, from full low collective to the maximum power output of the engines. The present system is not an improvement over a mechanical system. At the maximum rpm for flight (270 rpm or 105 percent), it was not possible to obtain full engine power and still maintain the selected rpm. This feature will be desirable during maximum performance operation. (B-11)

The power turbine speed governor (beep) switches must be utilized to overcome the deficiencies in the droop elimination system. Two switches are provided on the collective pitch head. The left switch controls both engines simultaneously and the right switch controls the number two engine. Operation is adequate; however, the rate of actuation is slow. Actuation time for the complete rpm range is approximately eight seconds. The time required to cover the small in-flight rotorspeed range is excessive. While hovering, this actuation rate changes the governed rpm approximately 4 percent (10 rpm) in four seconds. Due to normal delay in response time, this rate should be increased to provide a change in governed rpm of 4 percent (10 rpm) in at least three seconds from the beginning of actuation. (B 12)

Lag (temporary deviation of rotor rpm from the governed value during changes in power available) in the system should also be reduced. Rotor speed variations during collective pitch actuation may vary as much as 6 percent (15 rpm). Rotor speed variations should be limited to not more than 2 percent (5 rpm). (B 13)

Handling qualities through transition and climb are very good. Cyclic position gradient is positive with respect to speed and position changes are small with gain of airspeed. (There is no noticeable lateral or directional trim change through transition). Acceleration is rapid, and attitude change is not excessive. Three-per-rev vibrations of relatively high magnitude are encountered in the transition regime (from approximately 5-20 knots).

Airspeed is easily maintained in a climb and vibration levels are relatively low. At approximately 70 knots IAS, it was possible to match the maximum allowable torque limits on both transmission torque gages. This allowed maximum power to be utilized from both engines and is a desirable feature for this particular helicopter. The rate of climb is good.

DESCENT, APPROACH AND LANDING

Handling characteristics during descent, approach and landing are satisfactory.

Visibility during descents and approaches is slightly obstructed. The large instrument panel obstructs visibility forward and to the left; however, visibility laterally and between the directional pedals is adequate if a slight sideslip is utilized.

Vibration levels are slightly higher than encountered in level flight at the same airspeeds during descent, but are not excessive. At about 15 knots IAS during transition to a hover, three-per-rev vibration increases to an uncomfortable level and remains until a hover is established.

Control during approach is good. An outstanding improvement is the lack of a large control position reversal during tansition and flare to a hover, which is an objectionable characteristic of the H-21. Adequate control power is available to level the helicopter after a steep flare at low airspeed.

Since the helicopter hovers in a nose high attitude, the flare during transition is quite steep and will further reduce visibility over the nose of the helicopter. The flare attitude is not excessive from the pilot's compartment, but appears excessive from the cabin section. The steep flare may also place the main landing gear in close proximity to the ground without the pilot being aware of the critical clearance.

There is no lateral or directional change during transition and hover is easily established. Landing is accomplished simultaneously on the main wheels. When lowering the nose wheel to the ground, it is necessary to apply brakes to prevent the helicopter from rolling forward.

During shut down, the SAS should be turned off and the nose strut must be fully compressed. For safety during flight, the lower SAS will not turn off unless a microswitch on the nose strut is activated. If the lower SAS is not off on reduction of rotor rpm, low rotor speed will cut out the a.c. generators, the gyros will tumble, and the SAS will sense a rate of deviation and apply a control input to the rotor system. This results in the blades striking the droop stops. While not dangerous, the contact with the droop stops is disconcerting.

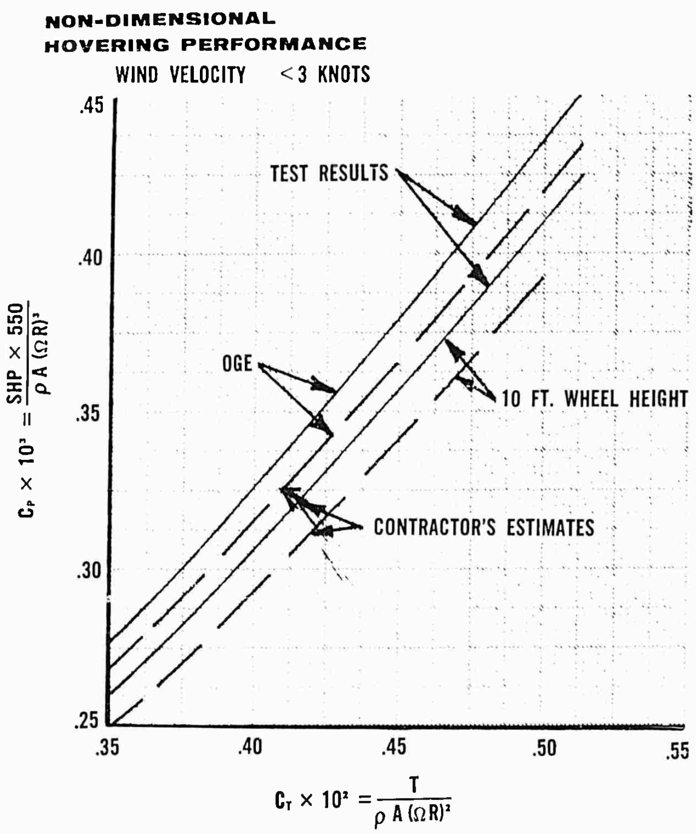
During an approach, the rotor rpm will normally overspeed about 10 rpm. Upon application of collective, the rpm will lower to its governed speed before the engines sense a requirement for additional power. Depending on the rate of collective pitch application, the rotor speed will lag 5 - 10 rpm lower than governed and result in a surge of power from the engines. To overcome this difficulty, excessive use of beep is necessary.

HELICOPTER PERFORMANCE

A limited amount of performance data was obtained during this test program. The areas of interest were hovering performance and range. Due to the short magnitude of the program it was impractical to obtain climb or descent performance and power available data. The power available data used to calculate the performance shown below was obtained from the contractor and is presented in Figure 15, Appendix I. It can be seen from this figure that for some standard day conditions the helicopter is limited due to the amount of torque the transmission is designed to absorb. These limits are discussed in detail in Appendix I; however, a transmission which does not limit engine power is desirable. (A-3)

Hovering Performance:

Hovering performance was obtained out of ground effect and in ground effect at wheel heights of approximately 4, 10, and 26 feet. This data is presented in non-dimensional C_P - C_T form in Figure 1, Appendix I. The figure below shows a comparison of the non-dimentional hovering performance obtained during this program with that estimated by the contractor out of ground effect and at a wheel height of 10 feet. For 270 rotor rpm and standard day conditions the difference in the predicted and test date would represent a hovering ceiling 1000 feet lower than Vertol estimates or approximately 350 pounds less weight lifting capability at the estimated ceiling. Actual hovering ceiling data was not obtained during this limited program.

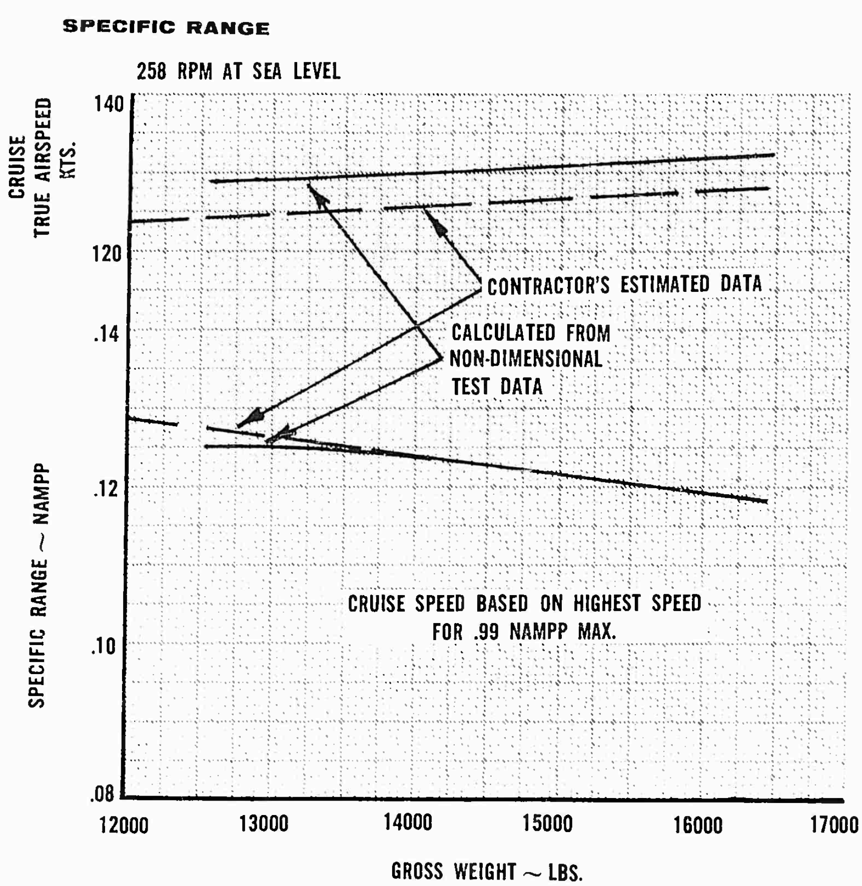


Level Flight Performance:

The maximum speed capability of the YHC-1A is very good. At 14,000 pounds at 1000 feet with 270 rpm the maximum speed of the vehicle is 147 knots true airspeed which exceeds the contractor's prediction of 137 knots. This speed is based on the transmission power limit of 2010 SHP at 270 rotor rpm with a 55-45 (aft-front) torque distribution. It should be noted that at 5000 feet and 270 rpm maximum power can not be maintained due to the droop eliminator (see Figure 5). This condition should be eliminated. (B-5)

Non-dimensional C_P, C_T, μ plots of the three speed power polars flown during this program are presented in Figures 3 and 4, Appendix I. This data agrees well with the contractor's prediction except at the low values of μ and the high values of C_T. For these conditions the actual power required is greater than the contractor's prediction.

A summary of the specific range of the helicopter at sea level utilizing 258 rotor rpm is shown in the figure below as a function of gross weight.



The specific range agrees well with the contractor's predictions; however, the recommended cruise speeds were considerably higher due to the flatness of the specific range curve, revealing that the actual range is fairly insensitive to speed. That data shown is based on the recommended cruise speed being the highest speed for 99 percent of the maximum nautical air miles per pound of fuel.

The specific range of the helicopter at higher altitudes (higher C_t's) will be slightly less than the contractor estimates due to the greater power required as explained above.

It is believed that the range of the helicopter could be substantially improved if retractable landing gear were incorporated. (B-3)

A calculated sea level range mission for this helicopter (crew of three) starting at the design gross weight of 15,550 pounds, a full load of fuel (2240 pounds) and oil (46 pounds), and 3690 pounds of cargo would be as follows:

Initial take-off gross weight	15,550 pounds
Warm-up and take-off fuel	41 pounds
Gross weight at start of cruise	15,509 pounds
Gross weight at end of cruise	13,492 pounds
Fuel reserve (10 percent of initial)	224 pounds
Average cruise speed	130.5 knots TAS
Time required for cruise	1.84 hours
Cruise rpm	258 rpm
Actual range	240 miles

It should be pointed out that this range is based on level flight data obtained in smooth air under ideal test conditions; however, helicopters flying under normal conditions (turbulent air) generally achieve 5 to 7 percent less range. Although sufficient data was not obtained to calculate a no cargo endurance mission, it should be pointed out that sufficient fuel should be made available for 4 hours endurance at sea level for these weight. (A-2)

Single Engine Operation:

Acceleration of the operating engine, following simulated failure of the second engine, is very rapid. The automatic and rapid assumption of the power loss by the good engine and the lack of trim or attitude change during power transients is a desirable feature. A loss of governed rotor rpm may be experienced if power available is not sufficient to meet power required, in which case, reduction of collective pitch to regain the desired rpm will be necessary. Handling qualities for single engine operation are identical to those for dual engine operation.

Airspeed Calibration:

The contractor airspeed calibration was spot checked in level flight and found to be quite accurate and acceptable.

The test aircraft was not equipped with a boom airspeed system and all airspeed indications were taken from the standard system which is located at the top of the cabin (see Figure 97, Appendix I). Flush static sources are located on the nose section of the fuselage. At the maximum rate of climb, a considerable position error is suspected. By maintaining attitude and reducing collective pitch to maintain level flight, indicated airspeed would rapidly reduce 10 to 15 knots. This effect was also noted in attempting to establish a climb in which the airspeed would indicate a rapid acceleration during the rotation of the fuselage to the climb attitude resulting in a higher indicated climb speed than desired. The opposite effect was noted during autorotation. These airspeed variations caused difficulty in establishing a desired airspeed during transition from one flight condition to another. Further investigation of this problem is necessary to substantiate the magnitude of position error and reduce this error. (B-14)

STABILITY AND CONTROL

In general, the stability and control characteristics of the YHC-1A are excellent. Ample dynamic stability is supplied by the SAS. The vehicle can be flown adequately under visual flight conditions to return to a landing field in the event of a complete SAS failure. Lateral and directional controllability (control sensitivity) is low but acceptable. Control harmony is good throughout the flight envelope and the aircraft level attitude at high speed is extremely desirable.

Static Stability:

The YHC-1A, regardless of c g location, exhibits positive speed stability above approximately 50 knots CAS with longitudinal cyclic and differential collective pitch (DCP) trim both operating. The aircraft is unstable from transition (approximately 20 knots) to maximum airspeed when both of these trim systems are inoperative. A comparison of the test results indicates that the cyclic and DCP trims have a great effect on speed stability while the c g location has a relatively slight effect. This analysis of speed stability was made from the trim curve data presented in Figures 16 through 21, Appendix I. (B-1)

Tests were also conducted at a forward c g to evaluate the static longitudinal stability at constant collective pitch and power turbine rpm settings. The variation of longitudinal control position with airspeed while displaced from a given trim airspeed indicated the same positive stability as discussed above and that the affects of power on longitudinal stability are negligible.

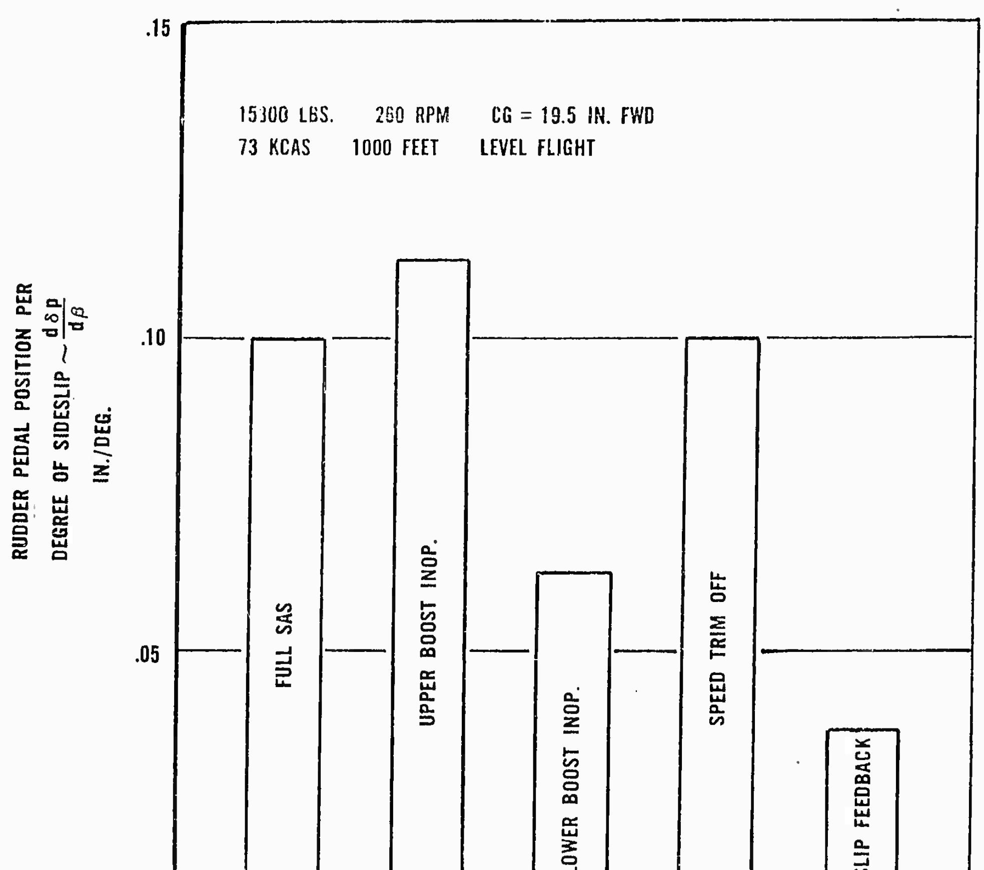
Speed stability during climb and autorotation were evaluated and found to be the same as for level flight, except for a small shift in the displacement of the longitudinal cyclic position.

The level attitude of the YHC-1A at high speeds is a very desirable characteristic from both a comfort and drag standpoint. Another desirable flying characteristic of this helicopter is the essentially neutral lateral cyclic and rudder pedal positions throughout the speed range. This is attributed largely to the cambered rear rotor pylon.

The YHC-1A exhibits positive static lateral and directional stability with full SAS operating in level and climbing flight. Static directional stability decreases with increasing rate of descent (see Figure 26, Appendix I) and becomes marginal at descent rates of 2000 feet per minute or greater. Near neutral stability at large sideslip angles is evident for the high rate of sink case as can be seen from Figure 35, Appendix I. Directional stability increased with increasing airspeeds. (A-4)

Comparisons were made of the apparent directional stability, $d\delta_p/d\beta$, with various modes of the SAS and boost systems inoperative. The figure below shows a comparison of these conditions.

STATIC DIRECTIONAL STABILITY



The test helicopter exhibited positive dihedral effect for all conditions tested, except at very large sideslip angles during autorotation when the dihedral effect became neutral.

Complete static directional stability test results are presented in Figures 25 through 39, Appendix I.

Dynamic Stability:

The short period stability of the YHC-1A with the SAS in operation is very good. This dynamic stability was evaluated by use of one inch pulse inputs with the SAS fully operational and with the SAS operating at half gain.

Normally, the dual SAS operates with both SAS operating at half gain. In the event of a failure of one system, the other system increases to full gain to provide the same degree of stability. However, damping with one SAS at half gain is adequate for mission completion.

The damping ratios and periods for the longitudinal and directional short period stability are presented in Figures 41 and 42, Appendix I. For normal operation, the damping ratio (ζ) for the longitudinal short period stability is approximately .45 and .7 for the directional short period stability which indicates a well damped oscillation. This degree of damping is quite adequate.

Time histories of the pulses performed at approximately 70 knots IAS are presented in Figures 43 -57, Appendix I.

The phugoid stability of the helicopter was investigated by trimming at 74 KIAS, then gradually reducing airspeed approximately 20 knots, and then slowly returning the cyclic to the trim position. A gentle phugoid exists which damps in approximately one to $1\frac{1}{2}$ cycles and has a period of approximately 70 to 75 seconds (see Figure 40, Appendix I). At speeds below 50 knots CAS, or with the longitudinal cyclic trim inoperative, the phugoid was divergent due to the negative speed stability.

Controllability:

The longitudinal control sensitivity of the YHC-1A is very good; however, the lateral and directional control sensitivities are marginal. The good dynamic stability as described in the previous section results in the lateral and directional sensitivities being acceptable; however, an increase in these sensitivities is desirable. The table below shows the variation in control sensitivity with airspeed with the SAS fully operational at a density altitude of 1000 feet and a normal gross weight and c g location.

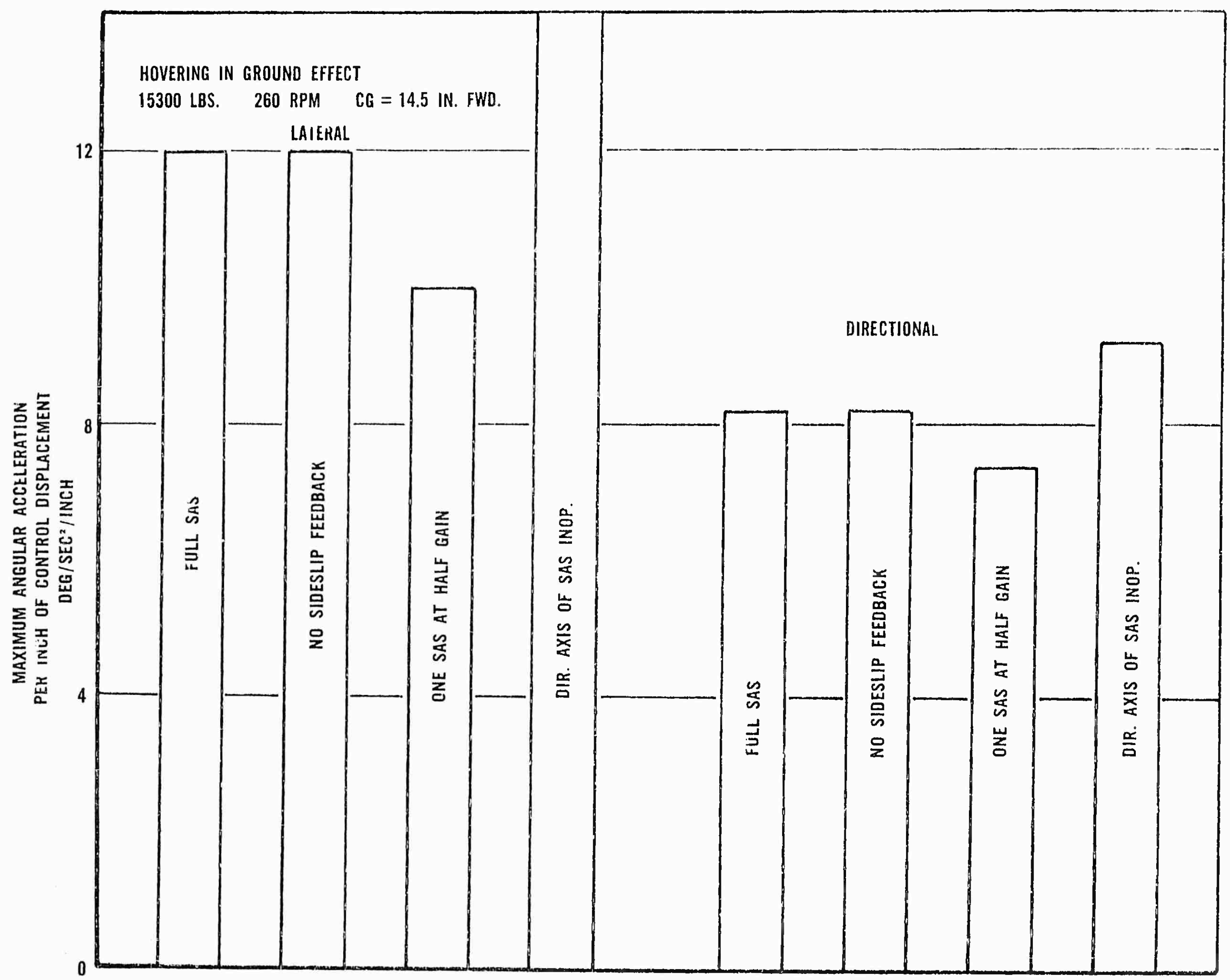
CONTROL SENSITIVITY

Deg/Sec^2/In.

Calibrated Airspeed - Knots	Pitch	Roll	Yaw
0	15.8	11.7	8.2
33	17.6	15.9	8.5
72	17.8	15.8	8.6
109	-	16.8	9.6

Control sensitivity tests were conducted about the lateral and directional axes for various SAS configurations during hover. The graph below shows the variation in the results.

CONTROL SENSITIVITY



Comparison of the YHC-1A control sensitivity with other helicopters which do not have stability augmentation systems is of interest. Such a comparison is made in the table below for the hover flight condition.

Helicopter	Control Sensitivity, Deg/Sec^2/Inch		
	Pitch	Roll	Yaw
YHC-1A (Full SAS)	15.8	11.7	8.2
H-21C	10.8	24.5	5.0
HU-1	8.0	24.5	32.6
YH-41 (Unmodified Version)	8.0	29.0	54.0

The time required to reach the maximum angular accelerations following a step input is of importance. For the YHC-1A this time delay is acceptable. During hover, the time to reach the maximum control sensitivity was approximately 0.2 seconds, increasing to approximately 0.5 seconds in forward flight about all axes.

The response of the helicopter to a step input in terms of the maximum angular velocities produced per inch of control displacement was also tested and analyzed. The table below shows results for level flight conditions with SAS fully operational.

	Response Deg/Sec/Inch		
Calibrated Airspeed - Knots	Max Pitch Rate	Max Roll Rate	Max Yaw Rate
0	6.6	4.2	5.2
33	5.5	3.8	4.0
72	5.9	3.8	4.2
109	-	3.9	4.5

These maximum angular velocities were obtained in 0.5 to 1.5 seconds.

Comparisons to other helicopters indicate that the response of the YHC-1A is less, except for the longitudinal axis. The following table is for hovering conditions.

Helicopter	Response Pitch Rate	Deg/Sec/Inch Roll Rate	Yaw Rate
YHC-1A (Full SAS)	6.6	4.2	5.2
HU-1	4.0	13.0	13.5
YH-41 (Unmodified Version)	7.0	17.0	31.0

A study should be made to determine the feasibility of increasing the roll rate and yaw rate per inch of stick or pedal displacement by a factor of three. (A-13)

Control sensitivity and response to step inputs are presented in Figures 58-87, Appendix I.

Flight Control System Tests:

Ground calibrations were conducted on the longitudinal and lateral cyclic control systems and the pedal system in order to ascertain the magnitude of the total forces and the frictional forces. Tests were conducted with both boost systems operative, and with the lower boost inoperative. The table below illustrates the results of these tests:

	Average Control Force Gradient lbs/in	Breakout Force lbs	Average Friction Band - lbs
Both Boost Systems Operating			
Longitudinal Cyclic	0.9	1.5	1.6
Lateral Cyclic	0.6	0.5	1.0
Pedal	2.0	6.0	0.8
Lower Boost Inoperative			
Longitudinal Cyclic	1.5	4.5	9.0
Lateral Cyclic	1.4	5.8	11.6
Pedal	3.0	10.5	14.0

The breakout forces in the cyclic control system are of approximately the same magnitude as the trim force gradient which results in an inability to trim out small forces in all regimes of flight. The average force gradient is excellent, but reduction of breakout forces is necessary to provide adequate trim capability. (B-2)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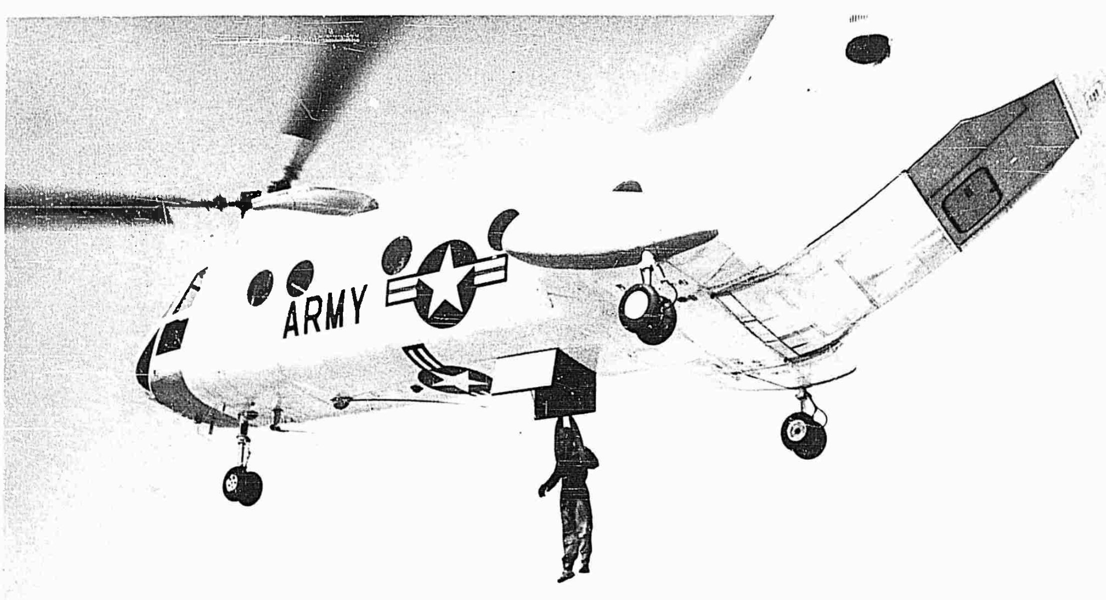

ARMY

Following a pedal pulse the helicopter will yaw, roll rapidly in the direction of the pulse, and then become divergent in roll oscillations.

A small lateral step input will also result in violent, divergent roll oscillations. The most aggravating reaction will be the continual deviations in yaw, while the most disconcerting reaction is the rapidity of roll.

It is possible to control the helicopter in climbs, descents, autorotation and hover. Control during maneuvering flight in mild turbulence is critical, and the violent stability reactions to small pulses and steps that are encountered makes it mandatory that any possibility of a simultaneous dual SAS failure during flight be eliminated. It is also desirable to provide a word warning and restrict the maneuvering envelope in the event of a failure of one SAS. (A-5)

In stabilized flight, loss of the dual SAS system may cause only minor control transients; however, if the dual SAS is lost during maneuvering flight, the SAS system will effectively introduce a step type control input to each control in the direction the helicopter is already moving. Depending on the degree and type of the maneuver, the helicopter may assume a dangerous flight attitude. The capability of the average helicopter pilot to make the proper control corrections is yet to be determined. Any situation that may lead to rapid sequence or simultaneous failure of both SAS systems is a safety of flight item.

At present, loss of rotor speed below the operating range will fail both a.c. generators, resulting in simultaneous loss of both SAS. An automatic emergency alternate a.c. source is desirable to operate the SAS in the event of primary a.c. source failure. (A-15) (B-29)

The SAS derives its signals from the gyros that operate the attitude indicators. Although the systems are considered independent, the remote indicating artificial horizon may be switched to derive its signals from the second system gyro. In the event an electrical malfunction of the remote indicator should result in loss of the gyro, selecting the second system position may result in loss of both gyro systems; therefore, loss of both stability augmentation systems.

The dependence of the SAS on the aircraft attitude sensing system is undesirable. Incorporation of separate rate gyros for the stability augmentation system is a necessity. (A-11)

<u>Boost Off Operation:</u>

The hydraulic control system has a three position control switch, "BOTH" "UPPER" and "LOWER". It is not possible to turn both boost systems off during flight. Flying qualities with both control boost systems inoperative were not tested.

The directional pedal force gradient is acceptable; however, breakout forces are excessive, especially in relation to the force required for full pedal deflection (13 pounds with a 6 pound breakout force). The breakout force should be reduced to within the 5 pound limit of MIL-H-8501. (B-2)

With the lower boost system inoperative, friction and breakout forces are excessive about all axes. In addition, the normal trim system will not trim out the cyclic control forces encountered in flight.

Control system dynamic response tests were conducted on the ground by displacing each control to full deflection and releasing it. The lateral cyclic and pedal control responses are essentially deadbeat; however, the longitudinal cyclic oscillated several times upon returning to neutral (see Figure 90, Appendix I).

The trim system control forces may be reduced to zero at any position of the controls by depressing the trim release button located on top of the cyclic control grip. The system is not capable of eliminating forces that are induced from other than the trim system itself, such as forces encountered with the lower boost system inoperative. The system is also deficient in eliminating small forces near trim due to the high breakout forces. Incorporation of a trim motor feature in conjunction with the force centering system to allow movement of trim center position to eliminate small residual forces, or utilization of the trim force gradient to overcome forces induced through the control system would be desirable. Replacing the automatic force centering system with a motor trim system is unsatisfactory since the variety of speeds, maneuvering requirements, and relatively large cyclic position changes through transition would necessitate undesirable trim requirement. (C-16)

SAS Off Operation:

Although the possibility of a double SAS failure is remote, a SAS off flight was made. SAS off operation of the YHC-1A is possible. Stability characteristics are marginal for flight in moderate turbulence and unacceptable for flight under instrument conditions. The vehicle could be safely returned to a landing field under visual flight conditions; however, all possibility of a dual SAS failure resulting from any single malfunction should be eliminated. (A-1)

The aircraft will pitch up (or down) and will continue to pitch up (or down) without apparent increase or decrease in pitching rate. Control power is sufficient to maintain a pitch attitude; however, the aircraft continually deviates from the desired attitude and continual longitudinal cyclic application is necessary to maintain an attitude.

The aircraft exhibits negative dynamic stability characteristics directionally and continually deviates from the desired heading. Continual one-half inch pedal applications are necessary to maintain a desired direction of flight.

Operating on both systems, control slop or delay in response is not encountered. When operating on "UPPER" boost only, an excessive static left lateral force is necessary to maintain the desired cyclic control position. Smaller static forces are encountered in forward longitudinal cyclic and left directional pedal. It is not possible to trim these forces to zero with the force trim system. Approximately one-half inch play is encountered in the cyclic control, and forces for actuation are excessive for precise control of the helicopter. Directional pedal forces are excessive for all maneuvers. There is no delay in control response, and the aircraft can be controlled adequately. However, the lateral cyclic forces preclude extended flight with the lower boost system inoperative.

With the "UPPER" control boost system inoperative, there is no increase in control forces. Low sensitivity is encountered in level flight within approximately ±1 inch of lateral and longitudinal cyclic control from trim. Lateral and directional stability is adequate, but the aircraft continually oscillates in pitch. If the cyclic control is held fixed, the longitudinal pitching oscillation increases, damps to a low value, and then begins to diverge again. This effect is caused by the response delay of the longitudinal SAS in conjunction with the designed boost control linkage end play.

Longitudinal pitching is the most undesirable effect encountered with the upper control boost system inoperative, and should preclude extended flight in turbulence. However, adequate control is available to safely fly, hover, and land the helicopter.

Sideward and Rearward Flight

Sideward and rearward flight were investigated to the maximum of the established flight envelope. One of the most noticeable flying quality improvements over previous tandem rotor helicopters is the lack of nose down pitching movement with tailwinds above 15-20 knots. Three hundred sixty degrees hovering turns were accomplished in 20 knot winds without adverse pitching effects or compromise in aft cyclic control travel. In rearward flight, small longitudinal pitching motions are induced by collective pitch application and result primarily in difficulty in establishing a constant speed. Above approximately 20 knots rearward flight, an additional effect was noted that was disconcerting and may complicate downwind hovering capability. The tail of the helicopter will deviate to the left, resulting in a lateral displacement to the left and a rolling moment to the right. Sufficient control power is available to overcome this displacement; however, its cause should be determined and eliminated. Control restrictions were not encountered and speeds in excess of 30 knots appear feasible.

Sideward flight to 25 knots was accomplished. Little apparent control travel was necessary and stability was very good. The lateral fuselage angle was small and speeds in excess of 30 knots appear feasible.

Variations in control positions and aircraft attitude during rearward and sideward flight can be seen from Figures 23 and 24, Appendix I.

VIBRATION

The vibration characteristics of the YHC-1A at high speed are the best of any helicopter tested by the AFFTC.

As expected with a three bladed rotor system, the frequency of the predominant vibration harmonic is three oscillations per revolution (three-per-rev). The amplitude of the one-per-rev and two-per-rev frequencies is essentially negligible as can be seen from Figures 88 and 89, Appendix I.

Vibration levels were measured in the longitudinal, lateral, and vertical planes on the cockpit floor and in the cargo compartment near the aircraft cg In general, the longitudinal vibrations are greater in the cargo compartment while the lateral and vertical amplitudes are approximately the same for the cargo compartment and cockpit. Near maximum airspeed, the three-per-rev vibration in the cargo compartment begins to increase while the vibration level in the cockpit remains constant. The only undesirable vibration levels exist during air taxi.

CONCLUSIONS

The YHC-1A represents an advance in the art of helicopter design due to its excellent handling characteristics, positive dynamic stability, and low vibration levels at high speed. The rear ramp loading facilities, two-engine reliability, excellent maintenance accessibility and high ratio of cargo volume to airframe volume with adequate weight lifting capabilities makes the YHC-1A potentially the most operationally suited of any modern helicopter tested by the AFFTC.

This helicopter has several other desirable features worth noting. The level fuselage attitude at cruise speed is desirable both from a comfort and drag standpoint. Excellent cockpit emergency exits are provided and the general cockpit layout and facilities are noteworthy. The rapid and automatic response of the second engine and lack of aircraft trim disturbances following an engine failure are outstanding. The essentially neutral lateral cyclic and pedal positions throughout the speed range of the helicopter are desirable. The lack of a nose down pitching moment with tail winds above 15 to 20 knots and nose up pitching moments during low airspeed flares are noticeable improvements over previous tandem rotor helicopters.

The major undesirable features of the YHC-1A include its dependence on the SAS for adequate stability and control, inability to utilize full engine power due to transmission torque limits, excessive rpm droop during power application which can lead to the loss of generator output and a dual SAS failure, slow beep trim rate, the location and operation of the rotor brake and parking brake, the lack of a manual fuel control shutoff, lack of control of the fuel boost pumps, and lack of engine anti-icing and fire extinguishing equipment.

Additional shortcomings include poor static directional stability during partial power descents and autorotations.

Although the probability of a complete SAS failure is remote, a SAS off flight was made. SAS off operation of the YHC-1A is possible; however, stability characteristics are marginal for flight in moderate turbulence and unacceptable for flight under instrument conditions in that the possibility of safely returning to the base is considered to be remote.

RECOMMENDATIONS

Although the YHC-1A will never be put into production, an advanced version of the helicopter, the Vertol 107 Model II, will be in production in the near future.

A. It is recommended that the following changes be made to the Vertol 107 Model II helicopter to increase operational suitability:

1. Eliminate all possibility of a dual SAS failure resulting from any single malfunction and in the event of a dual SAS failure, the helicopter should not be flown under instrument conditions. (page 28).

2. Provide sufficient fuel for at least four hours endurance. (page 19)

3. Provide a transmission which will not necessitate limiting engine power. (page 16)

4. Increase the static directional stability during partial power descents and autorotation. (page 21)

5. Provide a "SINGLE SAS" word warning signal. (page 7, 27)

6. Provide more clearance between the emergency "fluid shut-off" handles. (page 8)

7. Provide engine anti-icing equipment and engine fire extinguishing capabilities. (page 8)

8. Incorporate a manual fuel control shut-off available to the pilot and the copilot. (page 9)

9. Provide pilot control of the fuel boost pumps. (page 9)

10. Relocate the parking brake to the left of the pilot under the instrument panel. The word warning for the parking brake should be accurate. (page 8)

11. Provide separate rate gyros for the SAS rather than using the artificial horizon gyros and improve the erection system of the horizon gyros. (page 12, 27)

12. Provide separate rotating axles for the nose wheels to make turning easier. (page 10)

13. A study be made to determine the feasibility of increasing the roll rate and yaw rate per inch of stick or pedal displacement by a factor of three. (page 25)

14. Relocate all circuit breakers to the cockpit. (page 9)

15. Eliminate the loss of both generators when rotor rpm drops below the normal operating range. (page 27)

B. The following changes should be made for improved service use of the helicopter:

1. Provide positive speed stability over the entire speed range of the helicopter. (page 20)

2. Reduce the control system breakout forces about all axes. (pages 25, 28)

3. Provide retractable landing gear for increased performance. (page 13)

4. Incorporate an automatic load sharing device for engine power. (page 8)

5. Improve the droop eliminator so that maximum permissible power can be maintained at maximum rpm. (page 13)

6. Provide an "ENGINE LOW OIL PRESSURE" word warning signal. (page 7)

7. Improve the detents of the condition lever panel so that the wrong condition cannot be inadvertently selected. (page 6)

8. Increase the illumination intensity of the master warning light. (page 7)

9. Relocate the rotor brake adjacent to the power quadrant so that it is available to both the pilot and the copilot. Also insure that the rotor brake warning light functions accurately. (page 8)

10. Provide a slower acceleration schedule for the rotor when going from ground idle to flight idle position of the condition lever. (page 12)

11. Improve the droop elimination system to maintain a maximum droop of 1 percent rotor speed throughout the available rotor speed range. (page 14)

12. Increase the rate of actuation of the rpm beep switches from four to three seconds for a 10 rpm change from the beginning of actuation. (page 15)

13. Decrease the temporary deviation of rotor rpm from the governed value during changes in power from 6 percent rpm to 2 percent rpm. (page 15)

14. Eliminate the erratic airspeed system operation during climb and autorotation. (page 20)

15. Provide engine torquemeters and engine fuel flowmeters. (page 8)

16. Provide adequate head room and a back rest for the crew member's seat. (page 6)

17. Eliminate the flight idle condition lever position. (page 12)

18. Extend the cargo hoist fuselage doors opening to approximately 45 degrees beyond the vertical to prevent damage of the doors if the hoist or sling cable should strike them. (page 11)

19. Relocate the master, battery and generator switch panel to the center of the console for accessibility by the pilot. (page 7)

20. Re-label "SAS CONTROL" and "BOOST CONTROL" functional arrangements to prevent confusion. At present the "SAS CONTROL" label is too close to the boost control and the "BOOST" label is in smaller letters than the "CONTROL" label. These labels should be located above the appropriate control. (page 7)

21. Provide ramp actuation control switches in the cockpit. (page 7)

22. Utilize mechanical rudder pedal adjustments in place of the electrical system in order to reduce complexity and improve reliability. (page 9)

23. Provide a method of operating the exterior ramp opening without requiring auxiliary power or entrance to the helicopter. (page 5)

24. Redesign the cockpit windows so that they may be opened or closed with one hand and so that they may be left open during flight. (pages 6, 9)

25. Provide direct entrance to at least the pilot's seat. (page 5)

26. Improve the taxiing characteristics with the nose wheel on the ground. (page 10)

27. Eliminate the small lateral and longitudinal movements of the helicopter which are apparent when hovering over a spot. (page 14)

28. Provide a trim release switch to be used when precise hovering is required. (page 14)

29. Incorporate an auxiliary power supply to operate the fuel controls when the battery is dead and to operate the SAS in the event of a primary a.c. source failure. (pages 9, 27)

C. The following minor changes should be made to the cockpit equipment and facilities for improved operation:

1. Provide an "APU CONNECTED" word warning signal. (page 7)

2. Rearrange the instrument panel in accordance with MIL-STD-33572 (page 7)

3. Provide foot brakes for the copilot. (page 8)

4. Improve the operation of the forward cargo compartment door so that it will not be difficult to open from the outside and so that it will not slide out of its tracks and prevent locking. (page 5)

5. Relocate the light control panel to the left of the console. (page 7)

6. Provide adequate identification of circuit breakers without restricting visibility of the circuit breaker. (page 7)

7. Provide a warning light dim switch in accordance with HIAD. The present system dims when the pilot's flight instrument light switch is "ON". This should be a separate circuit, returning to bright upon interruption of electrical power. (page 7)

8. Reverse the ignition switch actuation to have the switch forward while in normal position in accordance with MIL-STD-250A, 24 July 1956, Paragraph 1.3.a(1). (page 7)

9. Provide lighting for the circuit breaker control panel. (page 7)

10. Provide a channel frequency card holder adjacent to the UHF and VHF radios. (page 7)

11. Increase the length of at least one cargo compartment interphone cord to make it possible for a crew member to be stationed outside the helicopter for fireguard duty during starting and to enter the cockpit without changing headsets. (page 11)

12. Provide a handhold and one step for entering the forward cargo compartment door. (page 5)

13. Reduce the force required to actuate the mirror extension and retraction button. (page 6)

14. Install a flat plate between the console and the collective pitch attachment bracket to be used as a step when entering and leaving the pilot's seat. (page 5)

15. Raise the entire pilot seat adjustment range. (page 9)

16. Incorporate a trim motor feature in conjunction with the trim centering system to allow movement of the trim center position to eliminate small residual forces or utilization of the trim force gradient to overcome forces induced through the control system. (page 28)

D. It is further recommended that a study be made to insure that the above mentioned deficiencies of the YHC-1A are not present in the YHC-1B helicopters.


ARMY


US ARMY 85514
ARMY

APPENDIX I

TEST TECHNIQUE AND DATA ANALYSIS METHOD

General:

The equations and procedures used to analyze the performance and stability of the helicopter and to refer test conditions to U. S. Standard Atmosphere conditions are briefly described.

Dimensional analysis of the major items affecting helicopter performance will yield two sets of dimensionless variables which may be used to present performance data in non-dimensional form. The C_P, C_T, μ method is used in this report. These variables are defined as follows:

$$C_P = \frac{SHP \times 550}{\rho A (\Omega R)^3}$$

$$C_T = \frac{W}{\rho A (\Omega R)^2}$$

$$\mu = \frac{V_T}{\Omega R}$$

Where:

SHP = Total engine output shaft horsepower
ρ = Air density - slugs/ft^3
A = Total swept rotor disc area - ft^2
Ω = Rotor angular velocity - radians/sec
R = Rotor radius - ft
W = Gross weight - lbs
V_T = True airspeed-ft/sec

Hovering Performance:

Hovering performance was obtained out of ground effect by free flight hovering at approximately 200 feet above the ground and in ground effect at wheel heights 4, 10 and 26 feet. A sling load technique was utilized for hovering in ground effect with strain gage instrumentation in the swing line to measure the load exerted on the external weight. At all times the maximum weight of the helicopter with the sling load suspended in the air did not exceed the established weight limit for these tests.

This hovering data is presented in non-dimensional form (C_P vs C_T) in Figure 1. The engine shaft horsepower was obtained by the following relationship:

$$SHP = \frac{(RHP + 30)}{.955}$$

Where 30 HP are required for accessories and a 4.5 percent power loss in the transmission is assumed. (RHP is obtained from torquemeters)

Level Flight Performance:

The basis for correction of level flight speed power data lies in the C_P-C_T method. Each speed power was flown at an approximate constant C_T. This involves increasing altitude as fuel is used. The data was corrected for C_P to an exact constant C_T as follows:

$$C_{P_s} = C_{P_t} + \frac{\Delta C_P}{\Delta C_T} (C_{T_s} - C_{T_t})$$

Where $\Delta C_P / \Delta C_T$ is the slope of the C_P vs C_T curve at constant μ and the subscripts s and t denote standard and test conditions respectively. Shaft horsepower standard was then calculated using a standard rotor speed.

Specific range (nautical air miles per pound of fuel) was determined for standard day conditions by computing a standard day fuel flow as follows:

$$W_{f_s} = W_{f_t} + \frac{\Delta W_f}{\Delta SHP} (SHP_s - SHP_t)$$

When $\frac{\Delta W_f}{\Delta SHP}$ is the slope of the corrected fuel flow curve (Figure 11).

Power Available:

Power available data was not obtained during this program; therefore, contractor data for power available was used to calculate the hovering ceilings.

The torque distribution between the two rotors was measured during level flight and hovering tests and found to vary from 61 percent - 39 percent to 50 percent - 50 percent with the aft rotor having the greatest percent. The forward and aft transmissions have each been qualified at 60 percent of 1700 HP at a rotor speed of 258 rpm. The limiting RHP as measured by the rotor shaft torque-meter is .975 x .60 x 1700 or 995 HP per rotor. The table below shows the calculation for power available as limited by the transmissions:

Percent Torque Distribution (Aft-Fwd)	60-40	55-45	50-50
Fwd RHP at 258 rpm	662	814	995
Aft RHP at 258 rpm	995	995	995
Total RHP	1657	1809	1995

$*SHP_{258}$	=	$\left(\frac{RHP}{.955} + 30\right)$	1765	1920	2110
SHP_{264}	=	$\frac{264}{258} \times SHP_{258}$	1810	1965	2160
SHP_{248}	=	$\frac{248}{258} \times SHP_{258}$	1699	1848	2025

*Assuming 30 HP for accessories, a 2 percent power loss in the mix box and a 2.5 percent power loss in the transmission.

Dynamic Stability:

The oscillations following a pulse input are well damped; therefore, the convenient method of determining a damping ratio by counting the number of cycles to damp to a fraction of the initial amplitude will be of no merit with this helicopter. Two other methods were used to give an approximate damping ratio.

1. The overshoot method: The damping ratio is determined in this method by comparing the percent of overshoot of the first peak to a second order linear system (Reference BuAer Report No. AE 61-4).

2. Time Rise Method: The damping ratio (ζ) is found in this method by the relationship of t_2/t_1 to ζ where t_1 is the time for the response to reach 20 percent of the steady state value and t_2 is the time to reach 74 percent of the steady state value. The accuracy of this method depends on how well the beginning of the response can be identified.

The periods were determined from the following relationship:

$$T_n = T_d \sqrt{1 - \zeta^2}$$

Where: T_n is the undamped natural period

T_d is the damped natural period

Static Directional Stability:

The notation used on Figures 25 and 26 are:

$\frac{d\phi}{d\beta}$ = slope of bank angle, sideslip curve. Apparent dihedral effectiveness parameter.

$\frac{d\delta_{pedal}}{d\beta}$ = slope of pedal position, sideslip curve. Apparent dihedral stability parameter.

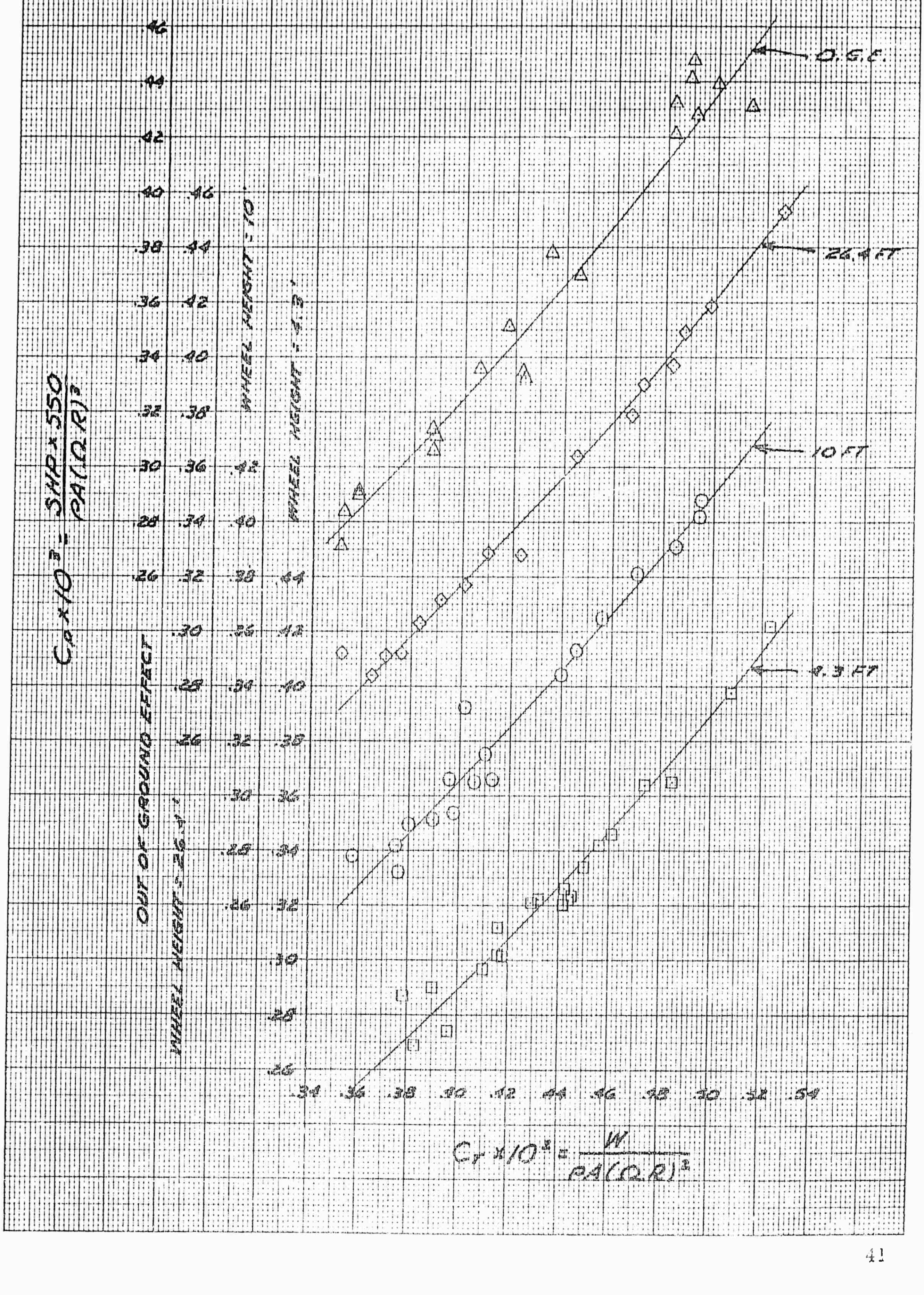
$\frac{d\delta_{lat\ cyclic}}{d\beta}$ = slope of lateral cyclic, sideslip curve. Apparent dihedral effective parameter.

FIGURE No 1

NON-DIMENSIONAL HOVERING PERFORMANCE

YHC-1A S/N 58-5514

WIND VELOCITY < 3 KNOTS



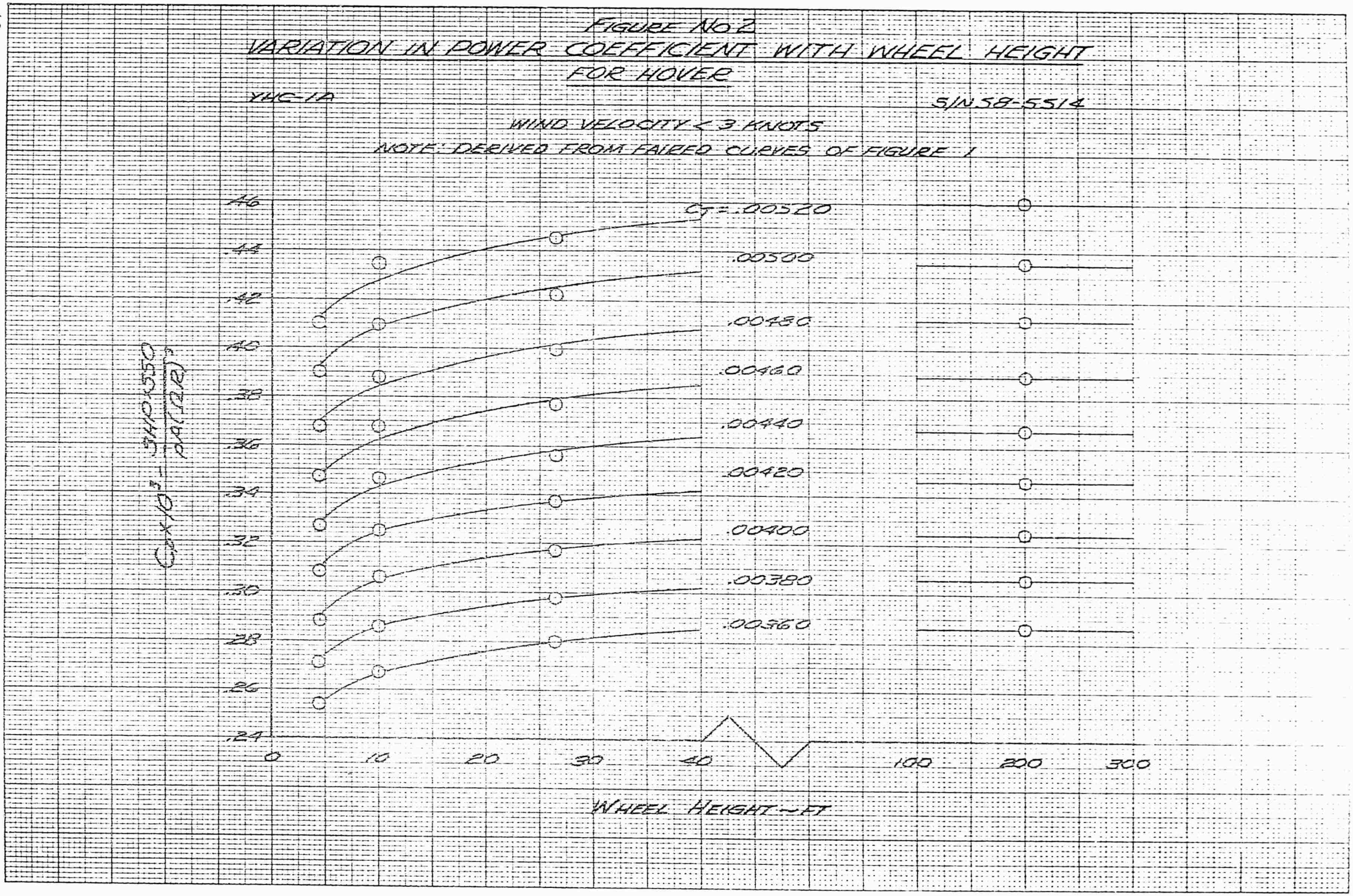


FIGURE NO 2
VARIATION IN POWER COEFFICIENT WITH WHEEL HEIGHT
FOR HOVER
YHC-1A
S/N 58-5514
WIND VELOCITY < 3 KNOTS
NOTE: DERIVED FROM FAIRED CURVES OF FIGURE 1
C_T = .00520
.00500
.00480
.00460
.00440
.00420
.00400
.00380
.00360
$C_P \times 10^3 - \frac{SHP \cdot 550}{\rho A (\Omega R)^3}$
.46
.44
.42
.40
.38
.36
.34
.32
.30
.28
.26
.24
0
10
20
30
40
100
200
300
WHEEL HEIGHT ~ FT

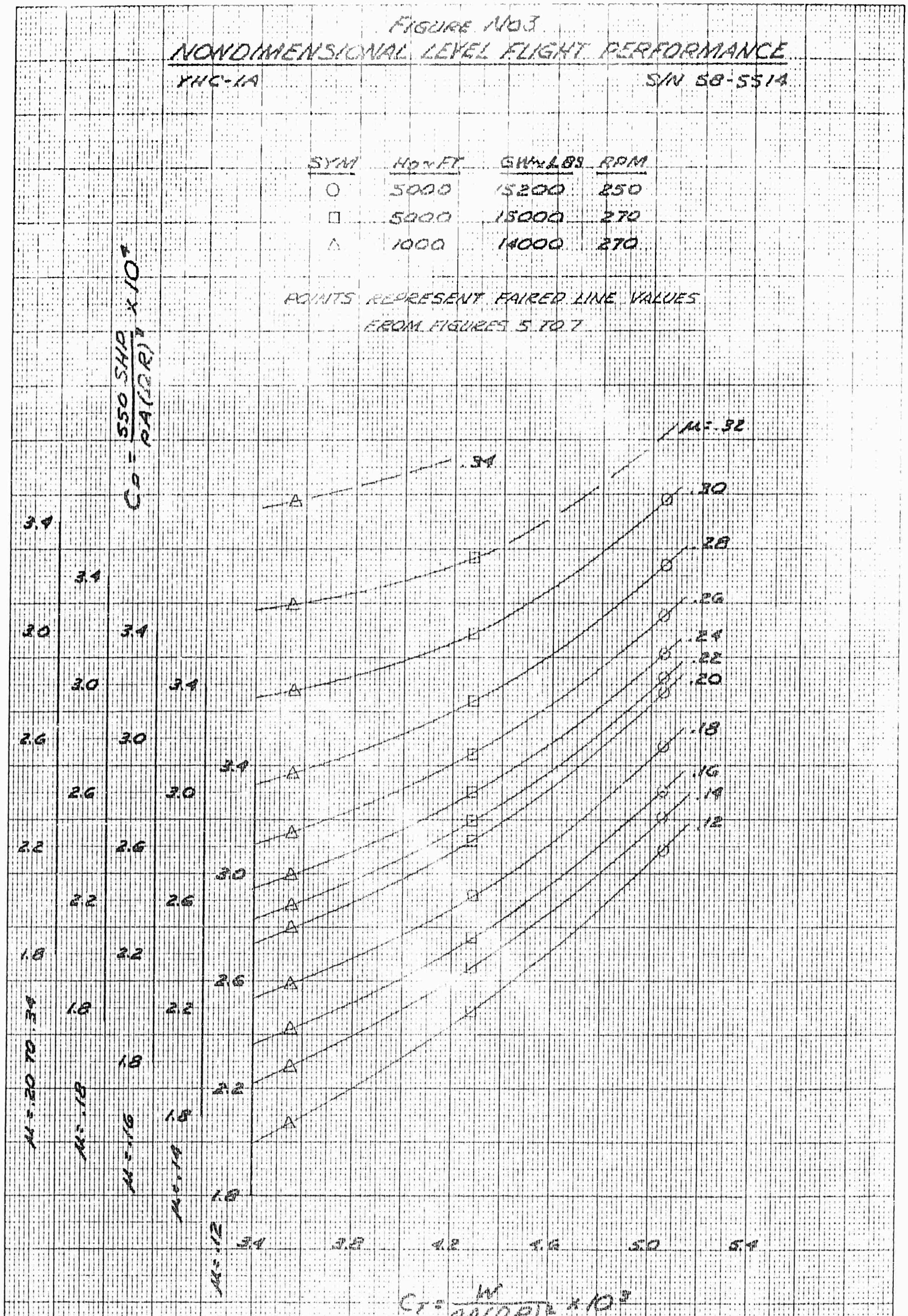


FIGURE No3
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

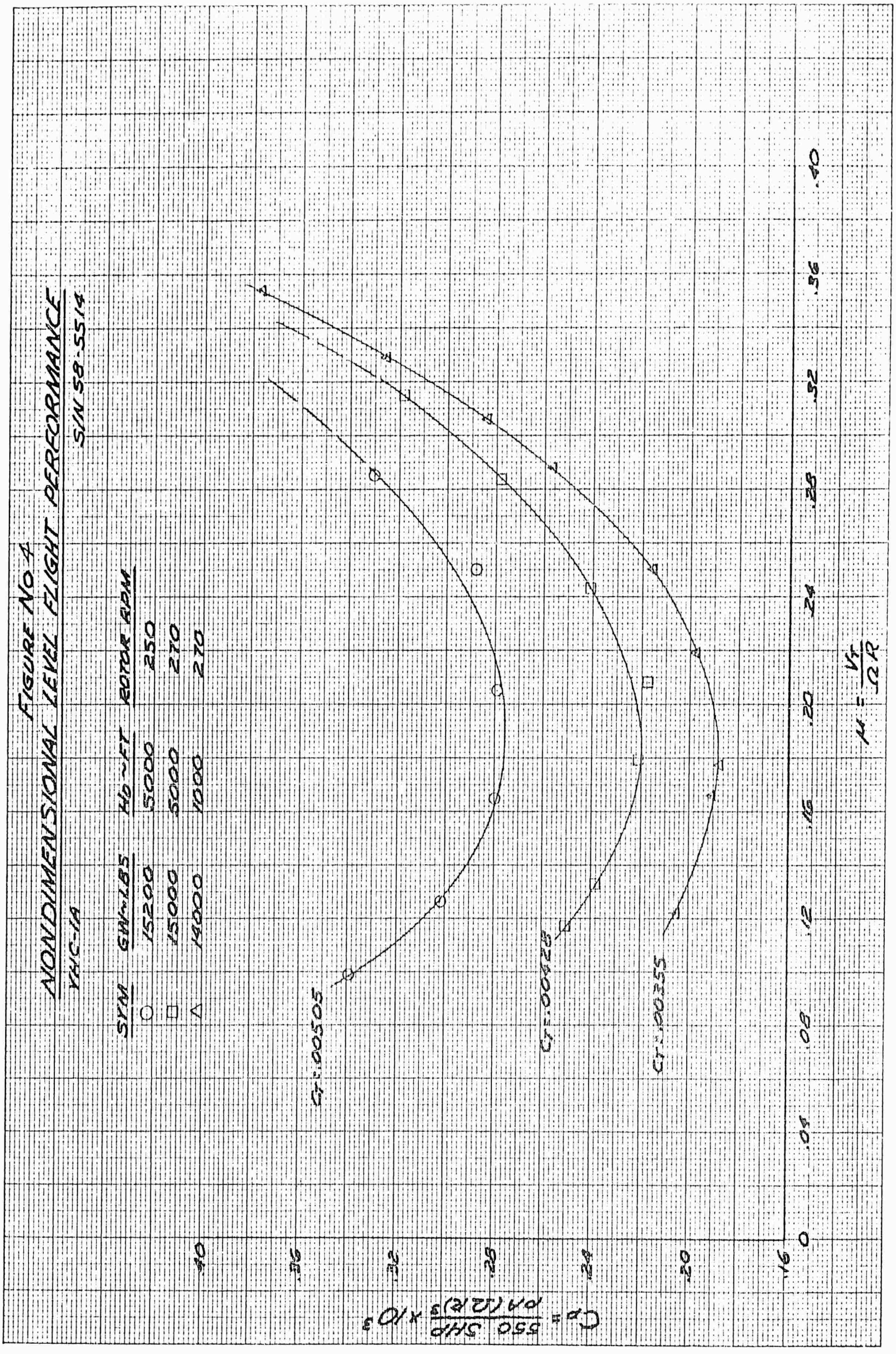


FIGURE No 4
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YHC-1A
S/N 58-5514
SYM GW~LBS HD~FT ROTOR RPM
○ 15200 5000 250
□ 15000 5000 270
△ 14000 1000 270
CT=.00505
CT=.00428
CT=.00355
0 .04 .08 .12 .16 .20 .24 .28 .32 .36 .40
$\mu = \frac{V_T}{\Omega R}$
.16 .20 .24 .28 .32 .36 .40
$C_P = \frac{550\ SHP}{\rho \pi R^2 (\Omega R)^3} \times 10^3$

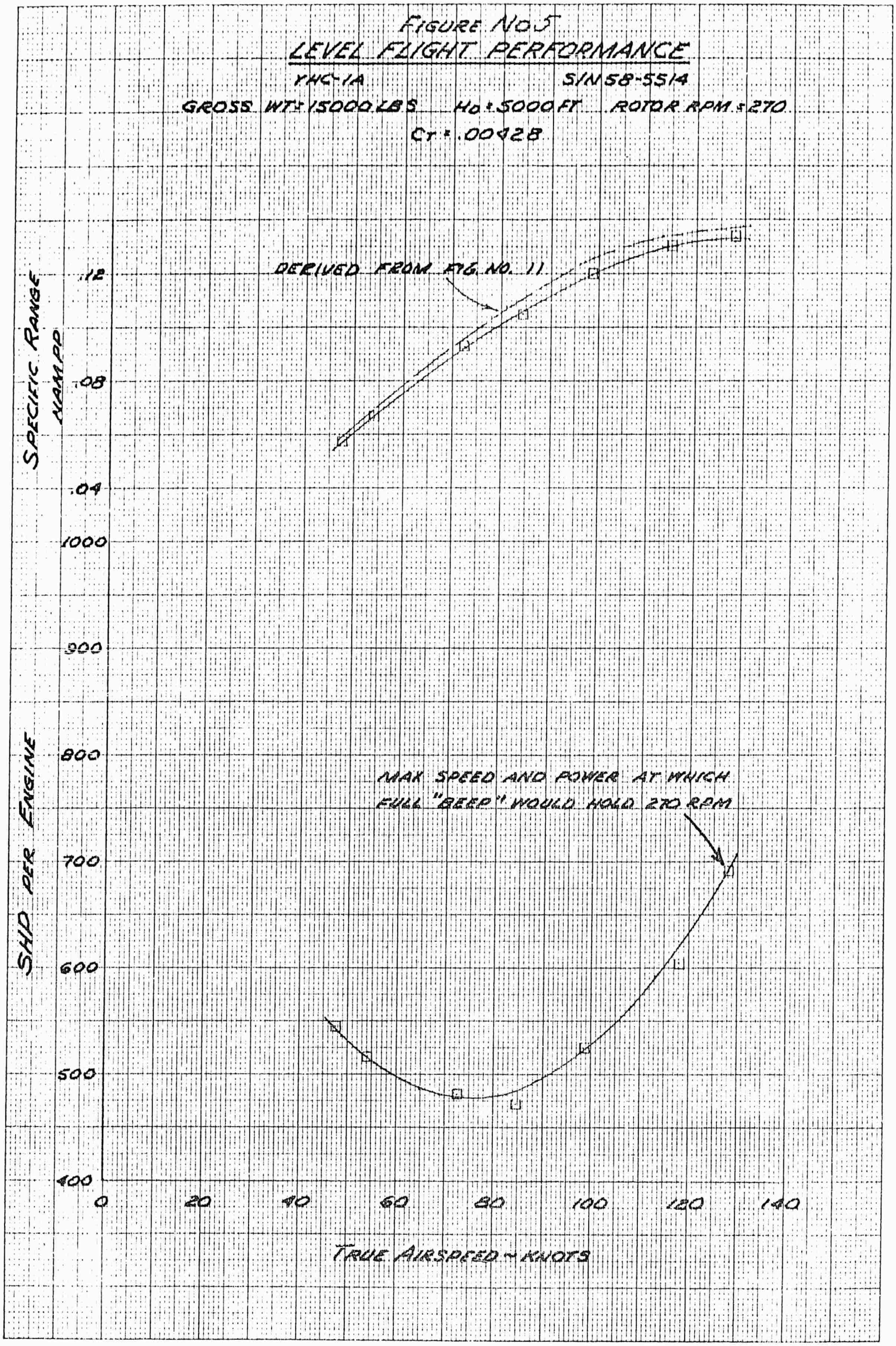


FIGURE NO 5
LEVEL FLIGHT PERFORMANCE
YHC-1A
S/N 58-5514
GROSS WT = 15000 LBS
Ho = 5000 FT
ROTOR RPM = 270
CT = .00428
DERIVED FROM FIG. NO. 11
SPECIFIC RANGE
NAM PP
.12
.08
.04
1000
900
800
700
600
500
400
SHP PER ENGINE
MAX SPEED AND POWER AT WHICH
FULL "BEEP" WOULD HOLD 270 RPM
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

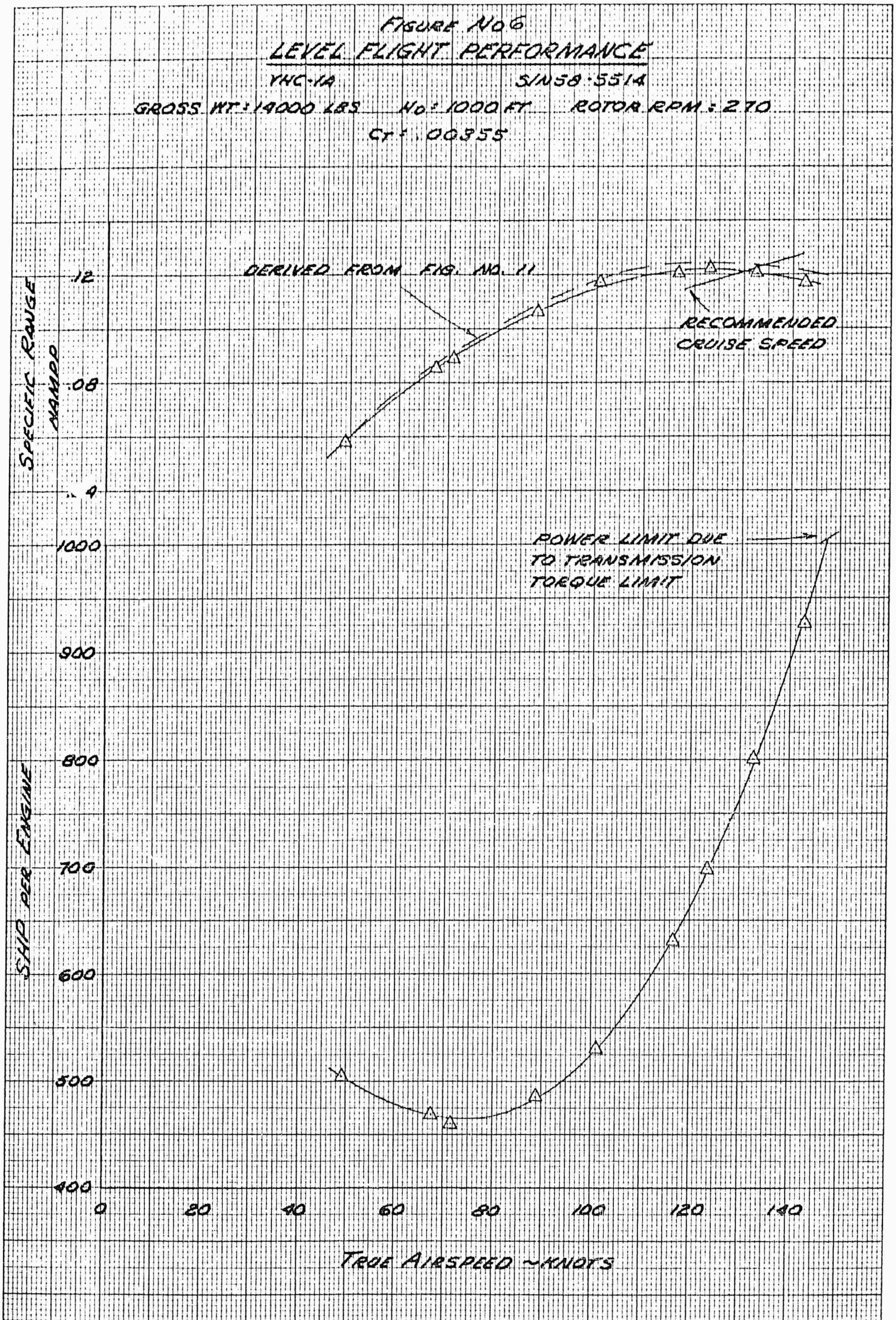


FIGURE NO 6
LEVEL FLIGHT PERFORMANCE
YHC-1A
S/N 58-5514
GROSS WT: 14000 LBS
Ho: 1000 FT
ROTOR RPM: 270
CT: .00355
DERIVED FROM FIG. NO. 11
RECOMMENDED CRUISE SPEED
SPECIFIC RANGE NAMPP
.12
.08
.4
POWER LIMIT DUE TO TRANSMISSION TORQUE LIMIT
SHP PER ENGINE
1000
900
800
700
600
500
400
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~KNOTS

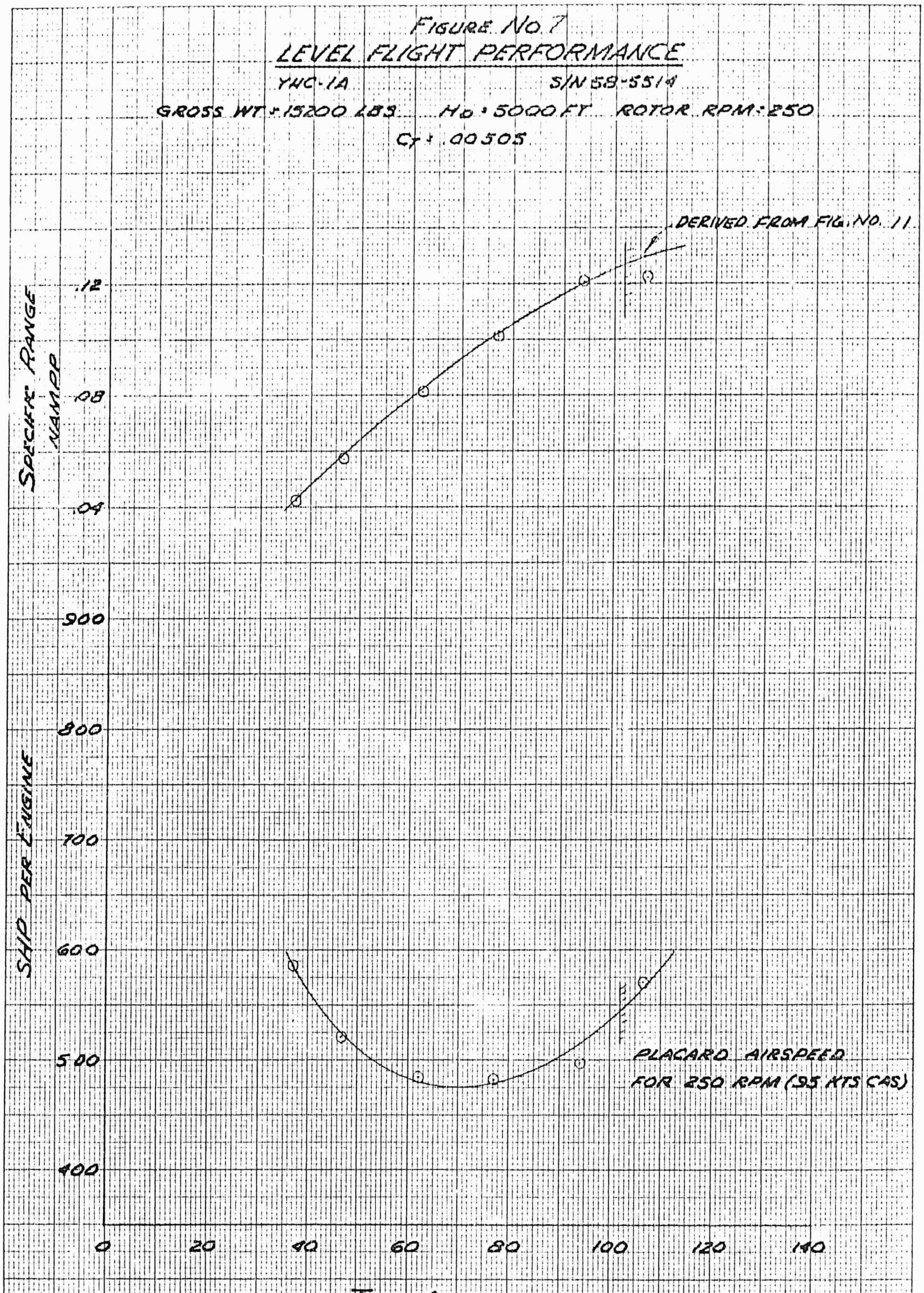


FIGURE NO. 7
LEVEL FLIGHT PERFORMANCE
YHC-1A
S/N 58-5514
GROSS WT = 15200 LBS.
Hb = 5000 FT
ROTOR RPM = 250
C_T = .00505
DERIVED FROM FIG. NO. 11
SPECIFIC RANGE NAMPP
.12
.08
.04
SHP PER ENGINE
900
800
700
600
500
400
PLACARD AIRSPEED FOR 250 RPM (95 KTS CAS)
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

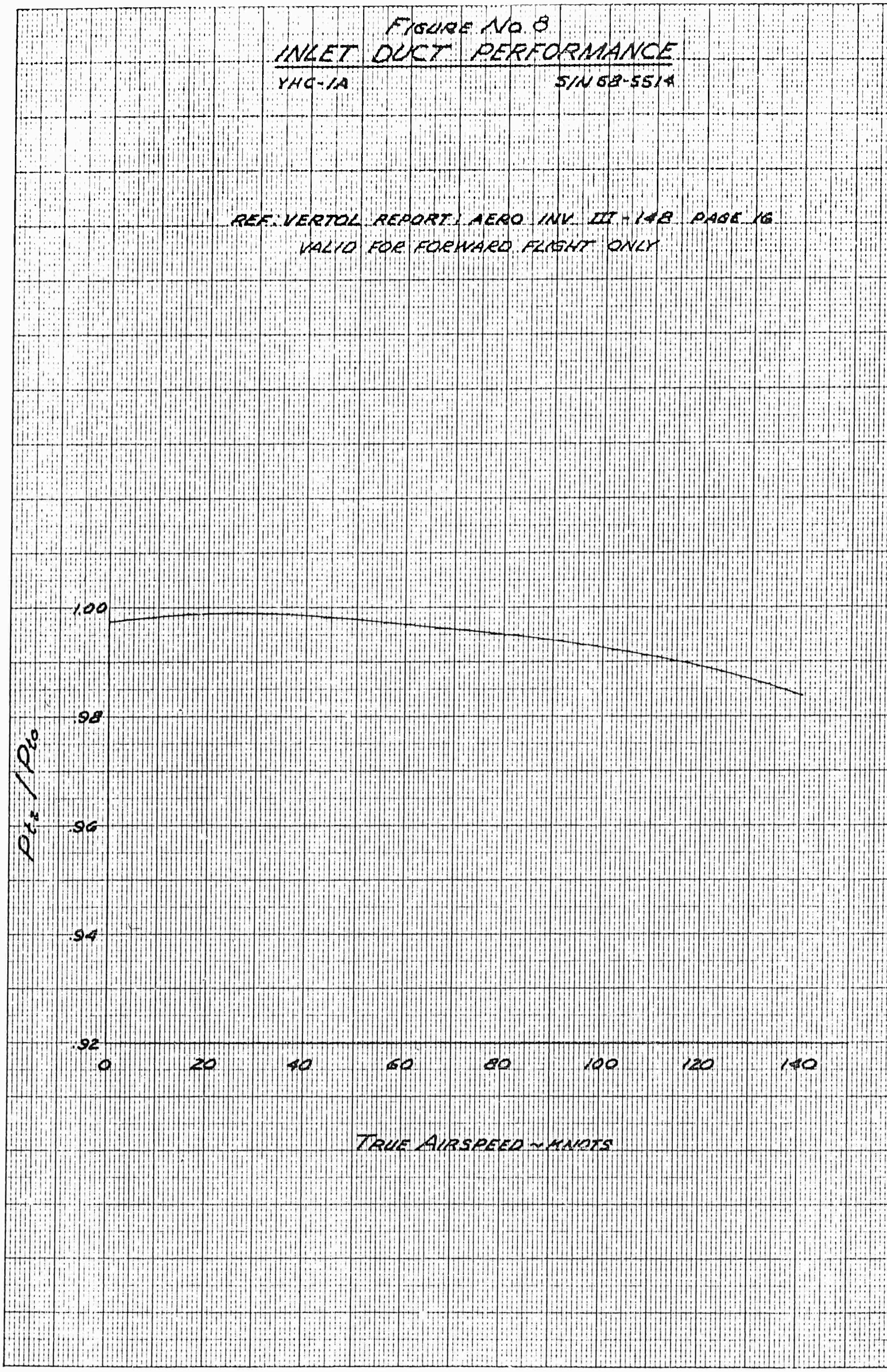


FIGURE No. 8
INLET DUCT PERFORMANCE
YHC-1A
S/N 58-5514
REF: VERTOL REPORT: AERO INV III-148 PAGE 16
VALID FOR FORWARD FLIGHT ONLY
1.00
.98
.96
.94
.92
Pt2 / Pt0
0
20
40
60
80
100
120
140
TRUE AIRSPEED ~ KNOTS

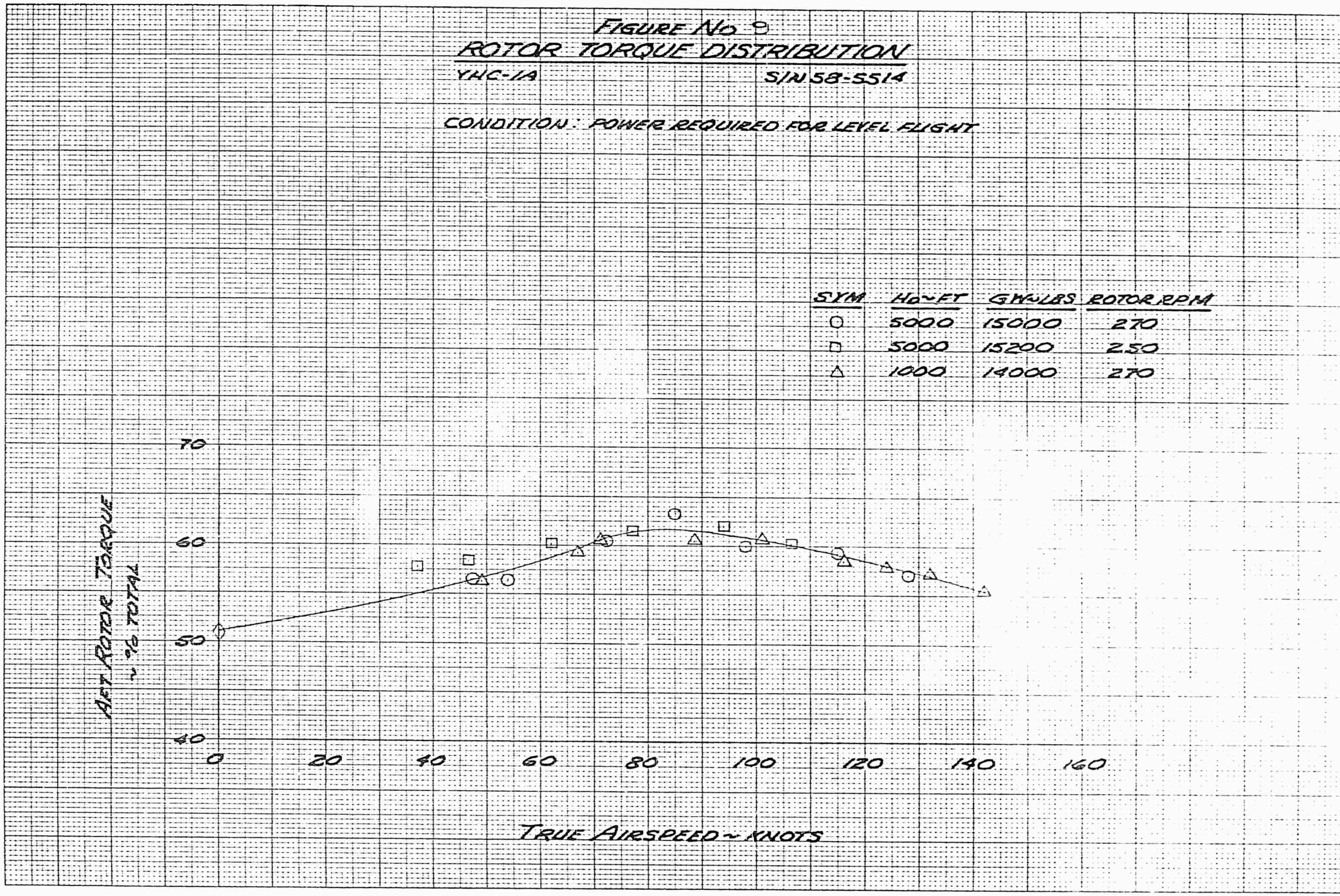


FIGURE No 9
ROTOR TORQUE DISTRIBUTION
YHC-1A
S/N 58-5514
CONDITION: POWER REQUIRED FOR LEVEL FLIGHT
SYM
H₀~FT
GW~LBS
ROTOR RPM
5000
15000
270
5000
15200
250
1000
14000
270
70
60
50
40
0
20
40
60
80
100
120
140
160
AFT ROTOR TORQUE
~ % TOTAL
TRUE AIRSPEED ~ KNOTS

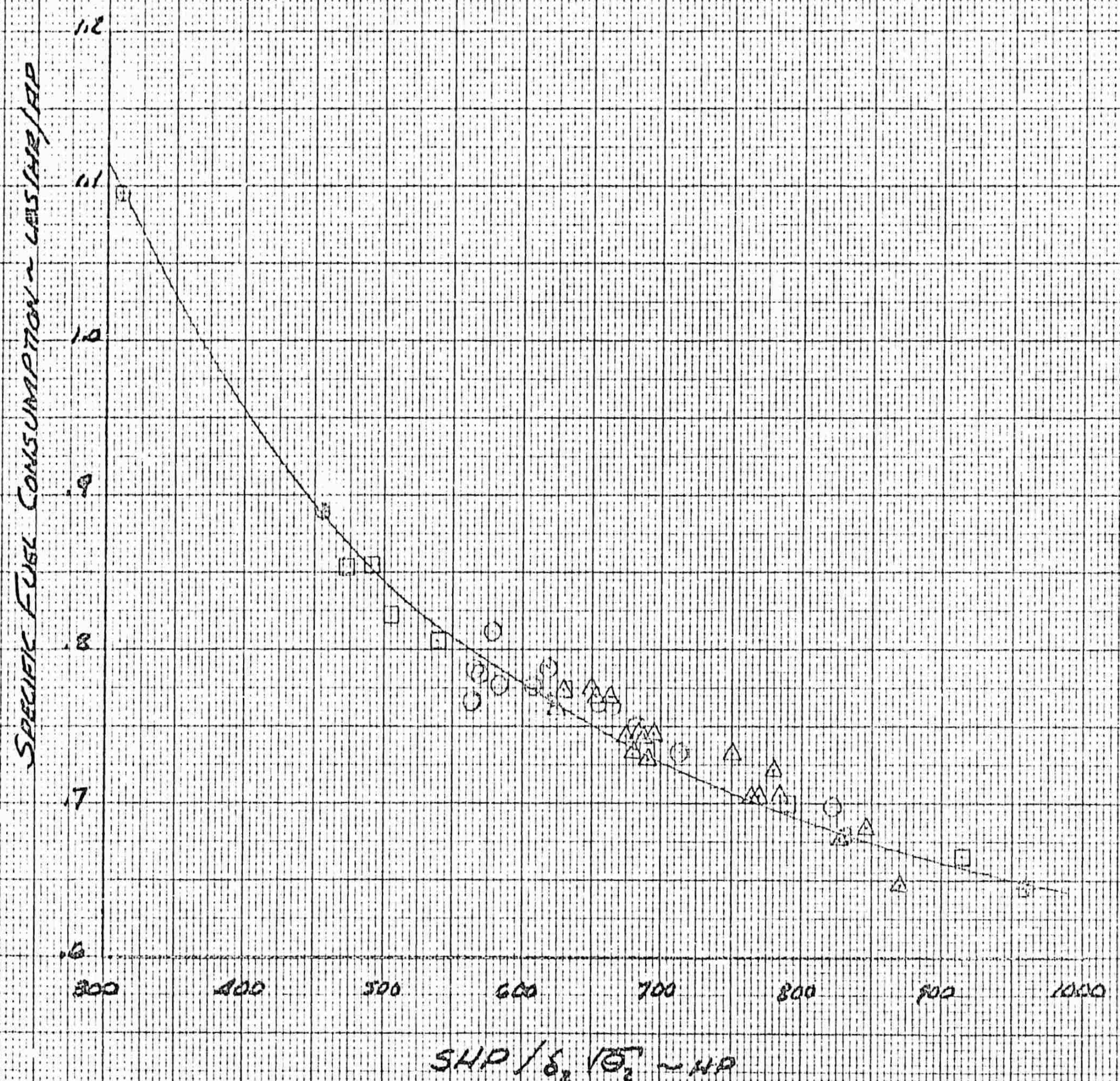
FIGURE No 10

SPECIFIC FUEL CONSUMPTION

YHC-1A S/N 58-5518

SYM	TEST	H_p ~ FT	ENG. S/N	
Φ	POST-TEST CALIB.	UNK.	200-117	200-120
□	SPEED POWER	1000	200-117	200-120
○	SPEED POWER	3000	200-117	200-120
△	HOVER OGE	200	200-119	200-118

NOTE: DATA AVERAGE OF TWO ENGINES



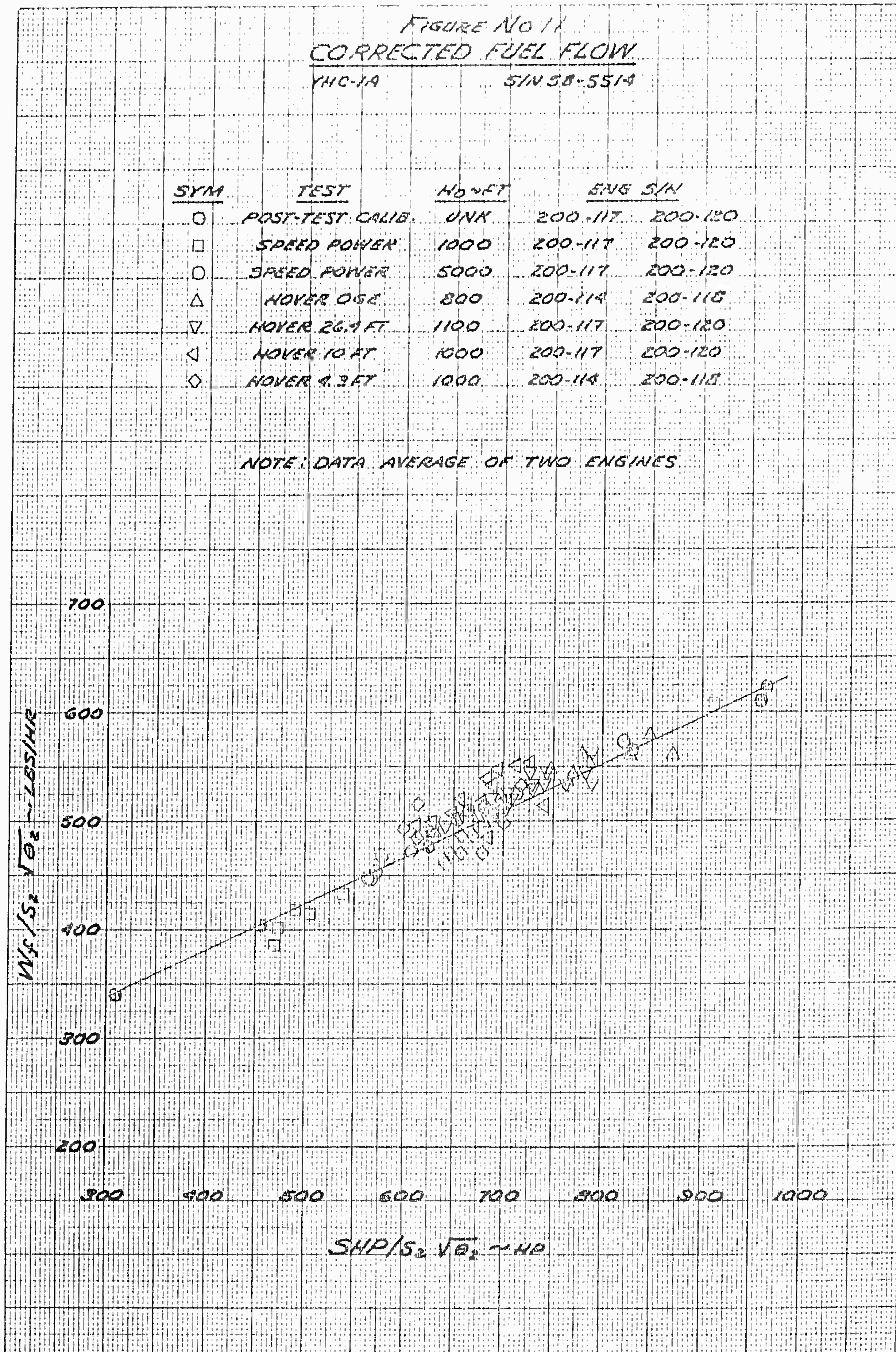


FIGURE NO 11
CORRECTED FUEL FLOW
YHC-1A
S/N 58-5514
SYM
TEST
H_P ~ FT
ENG S/N
POST-TEST CALIB. UNK 200-117 200-120
SPEED POWER 1000 200-117 200-120
SPEED POWER 5000 200-117 200-120
HOVER OGE 800 200-114 200-118
HOVER 26.9 FT 1100 200-117 200-120
HOVER 10 FT 1000 200-117 200-120
HOVER 4.3 FT 1000 200-114 200-118
NOTE: DATA AVERAGE OF TWO ENGINES
700
600
500
400
300
200
$W_f/S_2\sqrt{\theta_2}$ ~ LBS/HR
300
400
500
600
700
800
900
1000
$SHP/S_2\sqrt{\theta_2}$ ~ HP

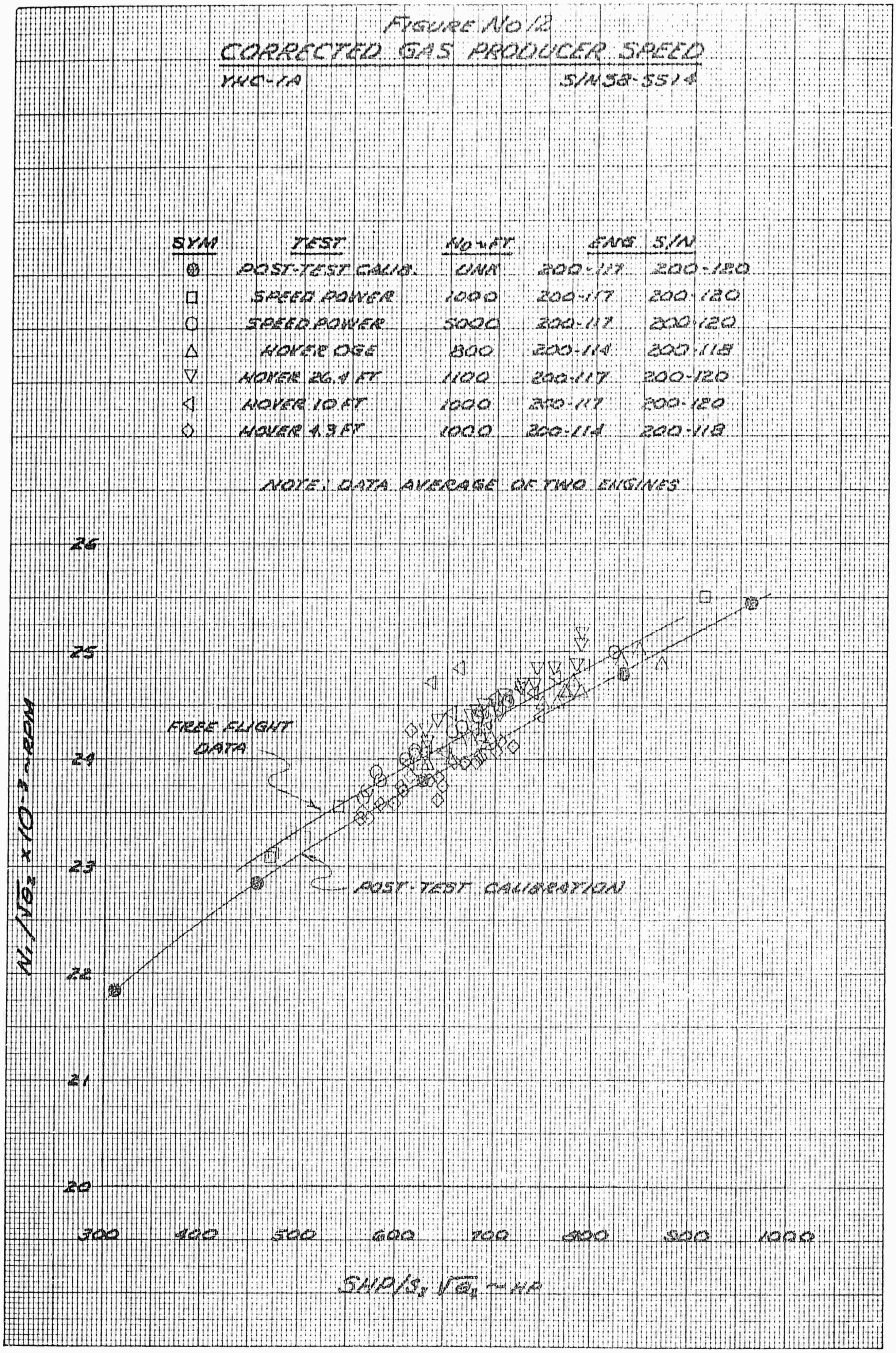


FIGURE No 12
CORRECTED GAS PRODUCER SPEED
YHC-1A
S/N 58-5514
SYM TEST No ~ FT ENG S/N
POST-TEST CALIB. UNK 200-117 200-120
SPEED POWER 1000 200-117 200-120
SPEED POWER 5000 200-117 200-120
HOVER OGE 800 200-114 200-118
HOVER 26.4 FT 1100 200-117 200-120
HOVER 10 FT 1000 200-117 200-120
HOVER 4.3 FT 1000 200-114 200-118
NOTE: DATA AVERAGE OF TWO ENGINES
FREE FLIGHT DATA
POST-TEST CALIBRATION
26
25
24
23
22
21
20
300
400
500
600
700
800
900
1000
$N_1/\sqrt{\theta_2} \times 10^{-3}$ ~ RPM
$SHP/\delta_2\sqrt{\theta_2}$ ~ HP

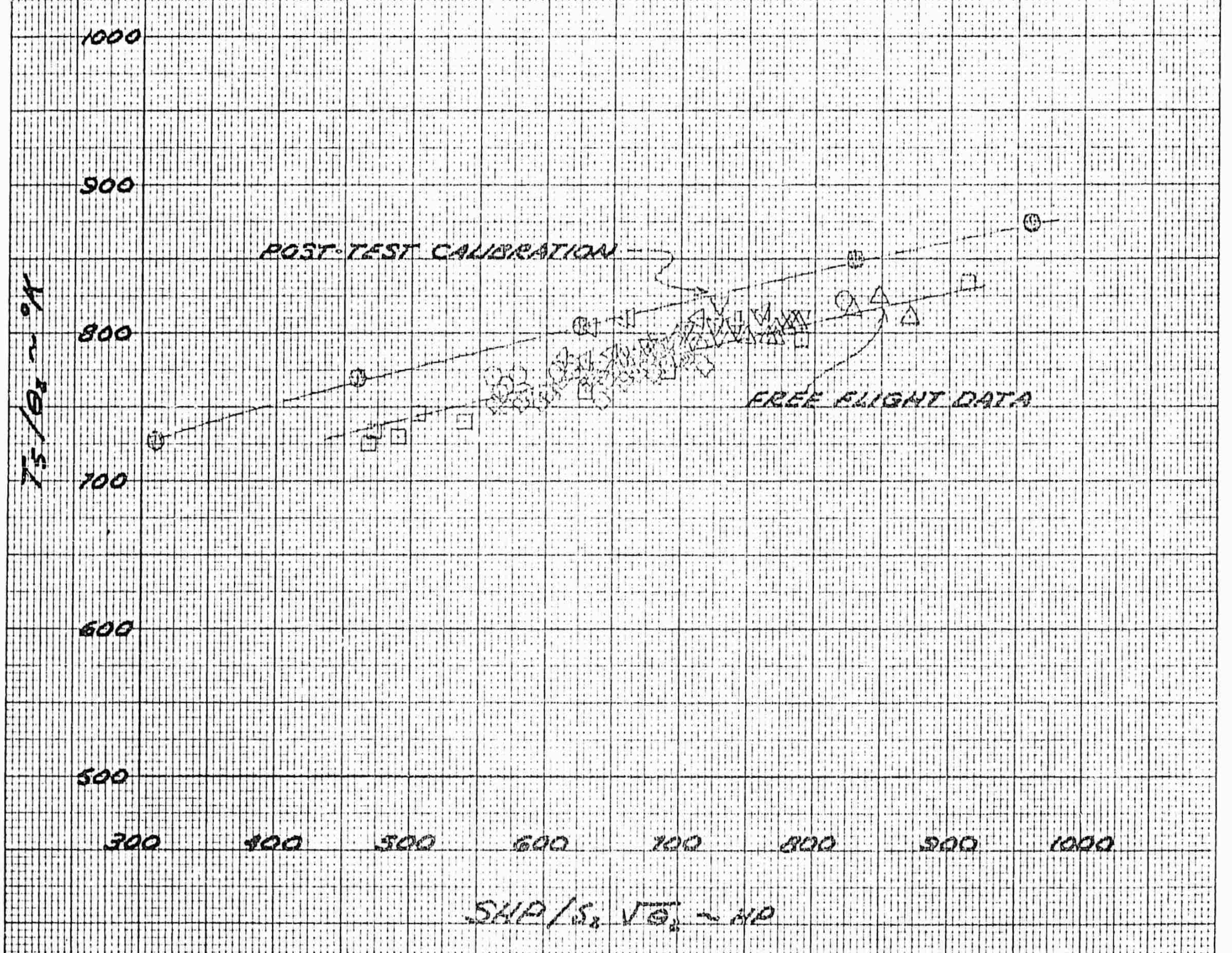
FIGURE NO 13

CORRECTED TURBINE INLET TEMPERATURE

YHC-1A S/N 58-5514

SYM	TEST	H_0 ~ FT	ENG S/N	
⊕	POST-TEST CALIB.	UNK.	200-117	200-120
□	SPEED POWER	1000	200-117	200-120
○	SPEED POWER	5000	200-117	200-120
△	HOVER OGE	800	200-114	200-118
▽	HOVER 26.4 FT	1100	200-117	200-120
◁	HOVER 10 FT	1000	200-117	200-120
◇	HOVER 4.3 FT	1000	200-114	200-118

NOTE: DATA AVERAGE OF TWO ENGINES



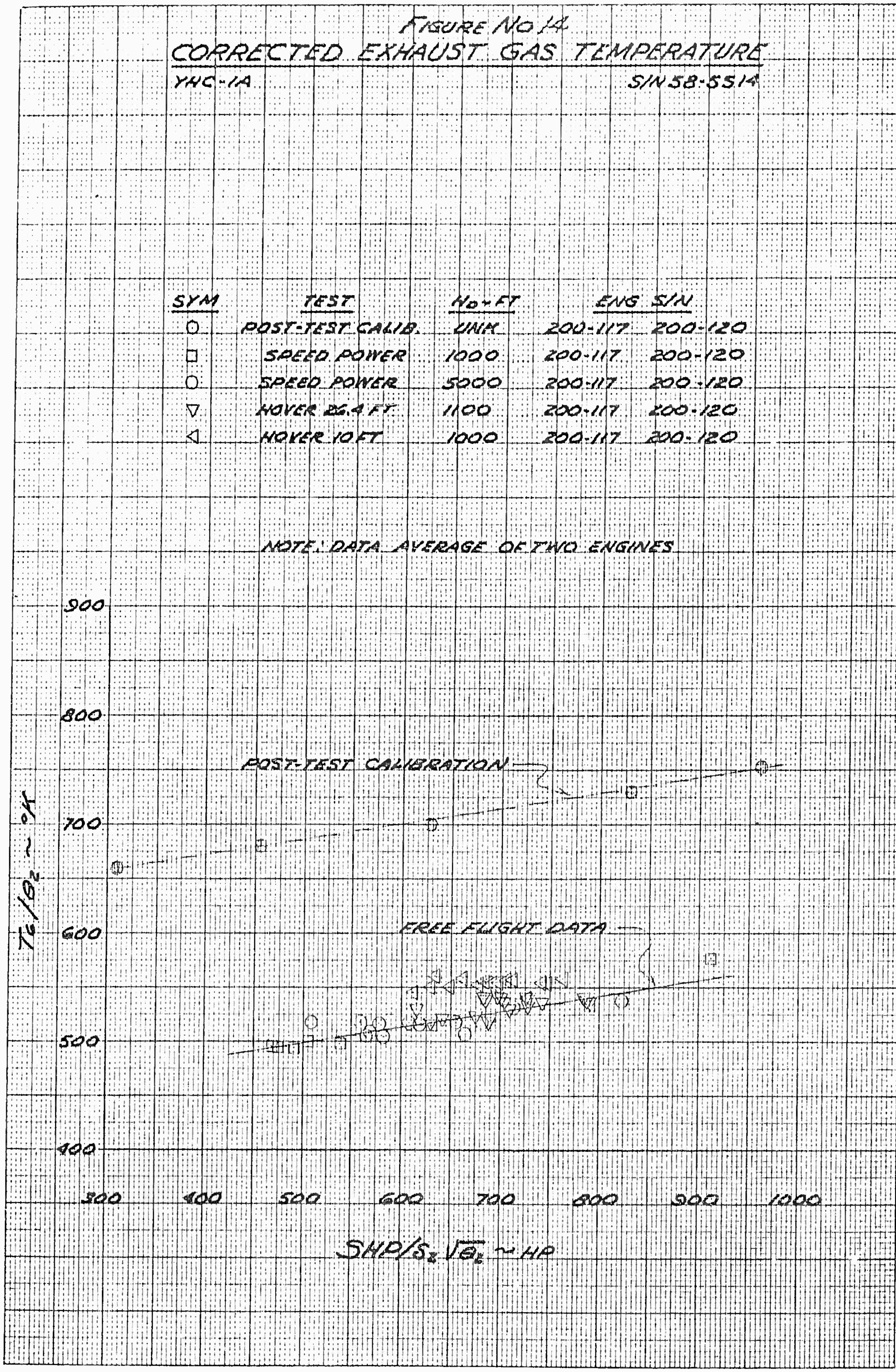


FIGURE NO 14
CORRECTED EXHAUST GAS TEMPERATURE
YHC-1A
S/N 58-5514
SYM TEST H_D ~ FT ENG S/N
○ POST-TEST CALIB. UNK 200-117 200-120
□ SPEED POWER 1000 200-117 200-120
○ SPEED POWER 5000 200-117 200-120
▽ HOVER 26.4 FT 1100 200-117 200-120
◁ HOVER 10 FT 1000 200-117 200-120
NOTE: DATA AVERAGE OF TWO ENGINES
POST-TEST CALIBRATION
FREE FLIGHT DATA
900
800
700
600
500
400
300
400
500
600
700
800
900
1000
T6/θ2 ~ °K
SHP/δ2√θ2 ~ HP

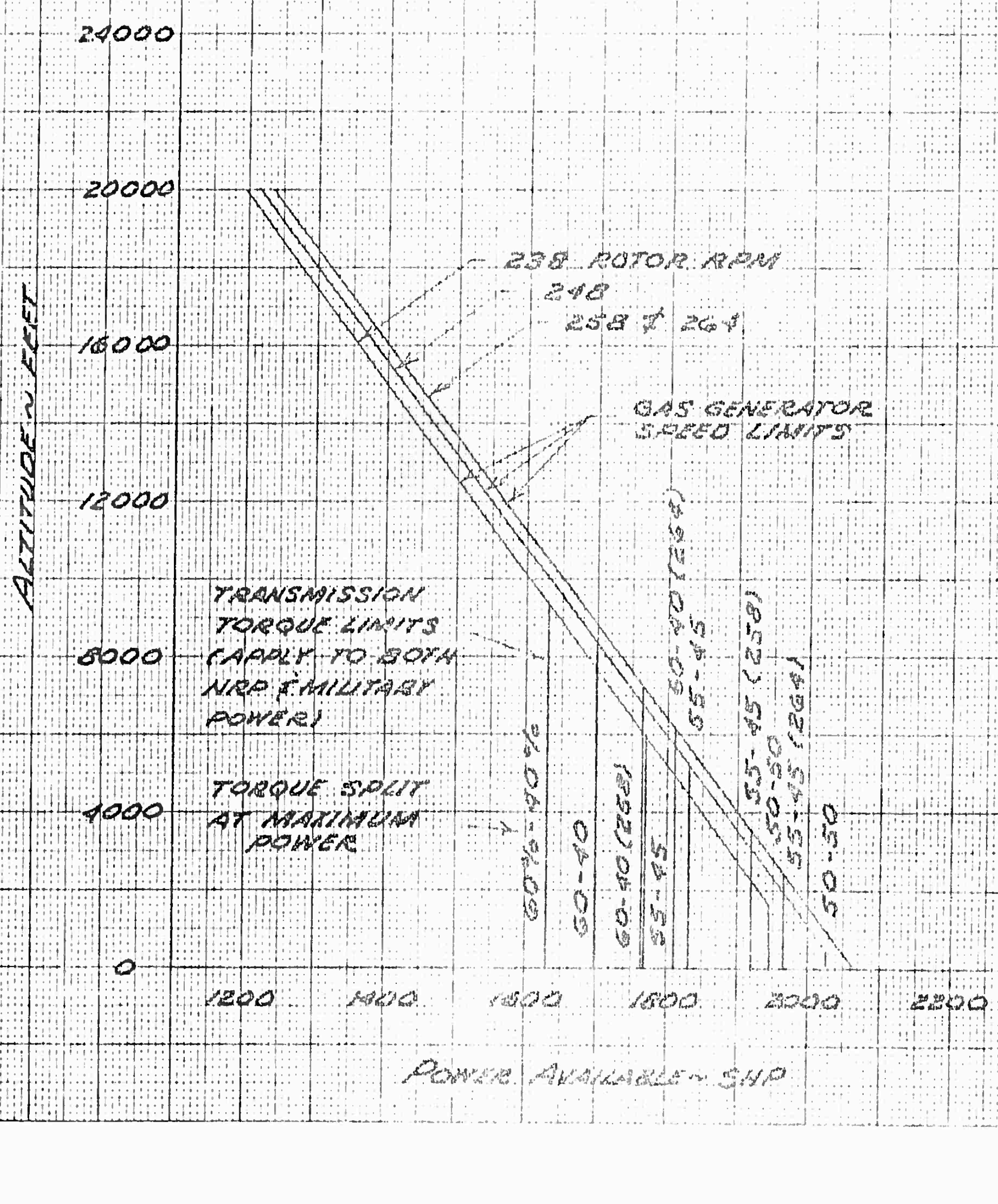
FIGURE No 15

POWER AVAILABLE PER AIRCRAFT VS ALTITUDE

YHC-1A S/N 58-5514

T 58-GE-6 SHAFT TURBINE ENGINE
MILITARY POWER
STANDARD ATMOSPHERE
HOVER

DATA OBTAINED FROM VERTOL



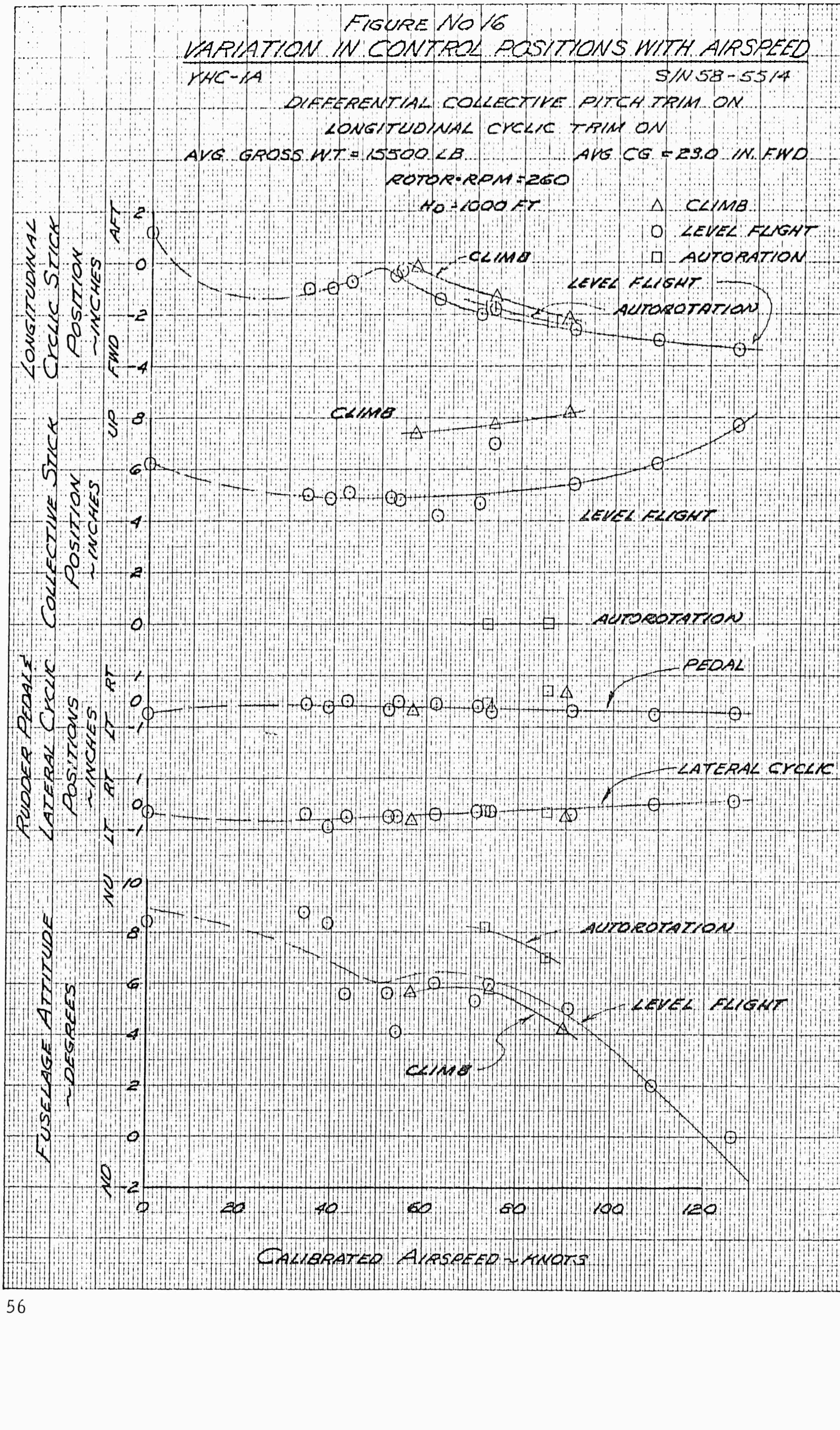


FIGURE NO 16
VARIATION IN CONTROL POSITIONS WITH AIRSPEED
YHC-1A
S/N 58-5514
DIFFERENTIAL COLLECTIVE PITCH TRIM ON
LONGITUDINAL CYCLIC TRIM ON
AVG GROSS WT = 15500 LB
AVG CG = 23.0 IN FWD
ROTOR RPM = 260
Ho = 1000 FT
△ CLIMB
○ LEVEL FLIGHT
□ AUTORATION
LONGITUDINAL CYCLIC STICK POSITION ~INCHES
AFT
FWD
CLIMB
LEVEL FLIGHT
AUTOROTATION
COLLECTIVE STICK POSITION ~INCHES
UP
RUDDER PEDALS LATERAL CYCLIC POSITIONS ~INCHES
RT
LT
PEDAL
LATERAL CYCLIC
FUSELAGE ATTITUDE ~DEGREES
NU
ND
0
20
40
60
80
100
120
CALIBRATED AIRSPEED ~ KNOTS

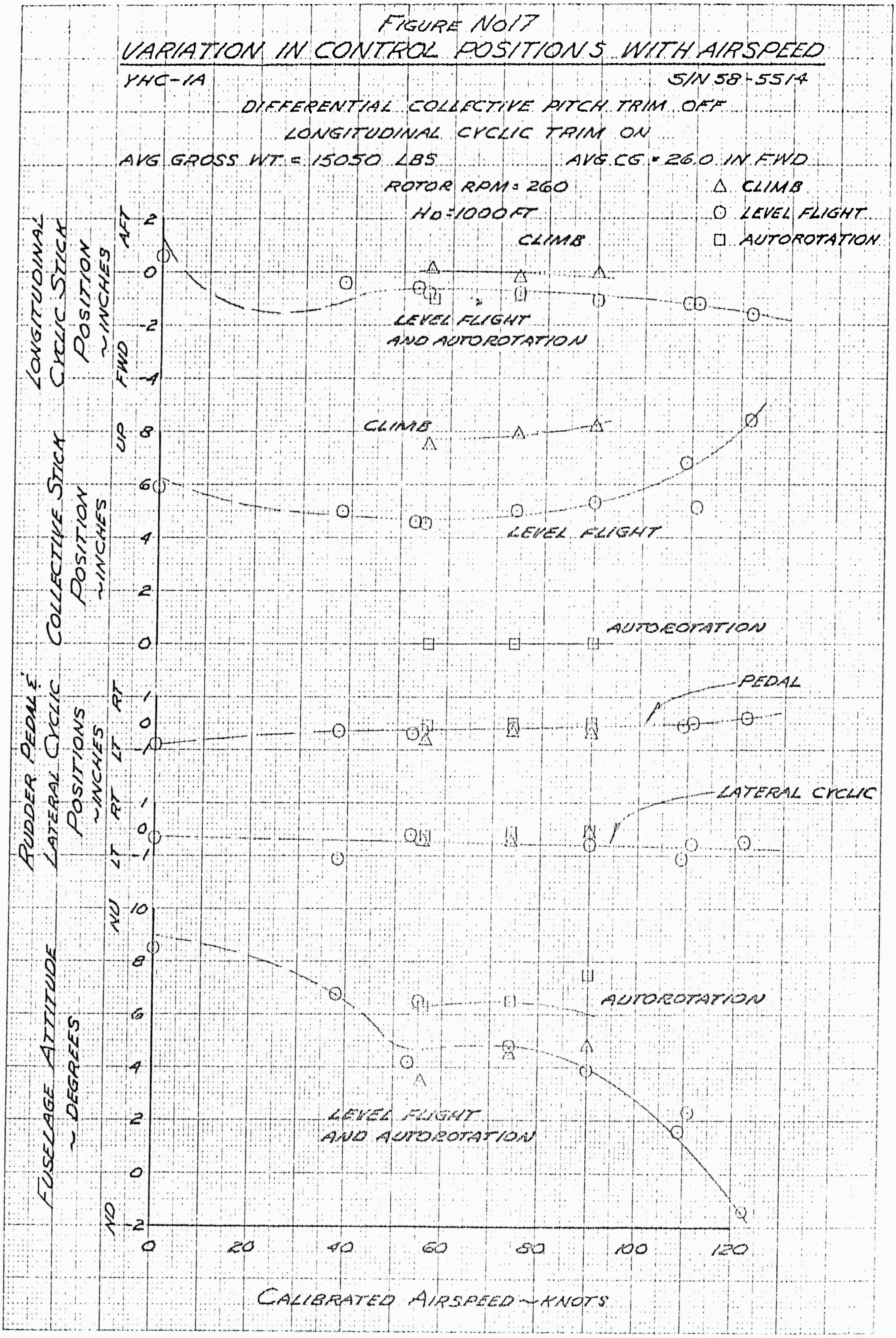


FIGURE No17
VARIATION IN CONTROL POSITIONS WITH AIRSPEED
YHC-1A
S/N 58-5514
DIFFERENTIAL COLLECTIVE PITCH TRIM OFF
LONGITUDINAL CYCLIC TRIM ON
AVG GROSS WT = 15050 LBS
AVG CG = 26.0 IN FWD
ROTOR RPM = 260
H_D = 1000 FT
△ CLIMB
○ LEVEL FLIGHT
□ AUTOROTATION
LONGITUDINAL CYCLIC STICK POSITION ~INCHES
AFT
FWD
CLIMB
LEVEL FLIGHT AND AUTOROTATION
COLLECTIVE STICK POSITION ~INCHES
UP
CLIMB
LEVEL FLIGHT
AUTOROTATION
RUDDER PEDAL & LATERAL CYCLIC POSITIONS ~INCHES
RT
LT
PEDAL
LATERAL CYCLIC
FUSELAGE ATTITUDE ~ DEGREES
NU
ND
AUTOROTATION
LEVEL FLIGHT AND AUTOROTATION
CALIBRATED AIRSPEED ~ KNOTS

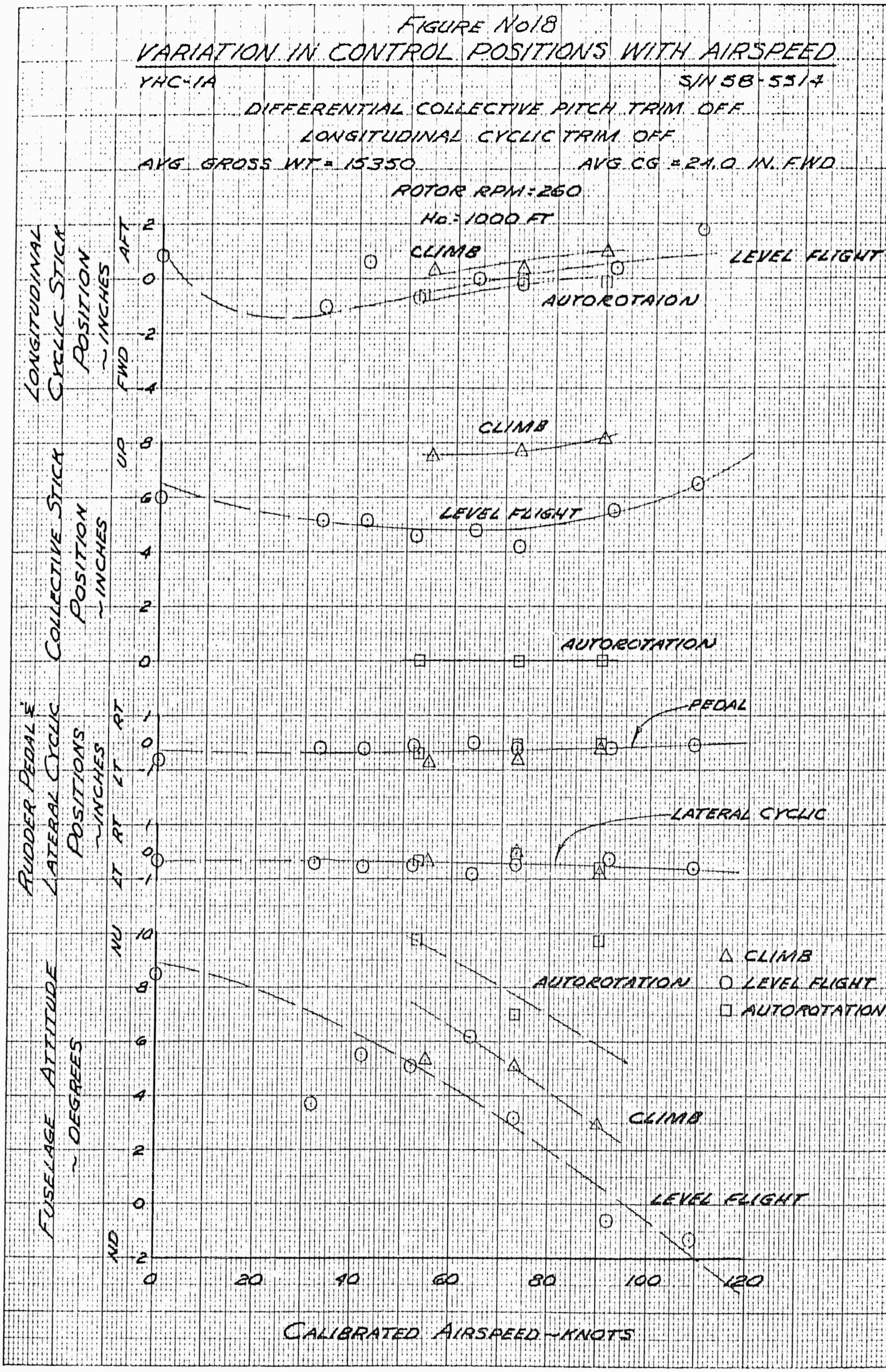


FIGURE No18
VARIATION IN CONTROL POSITIONS WITH AIRSPEED

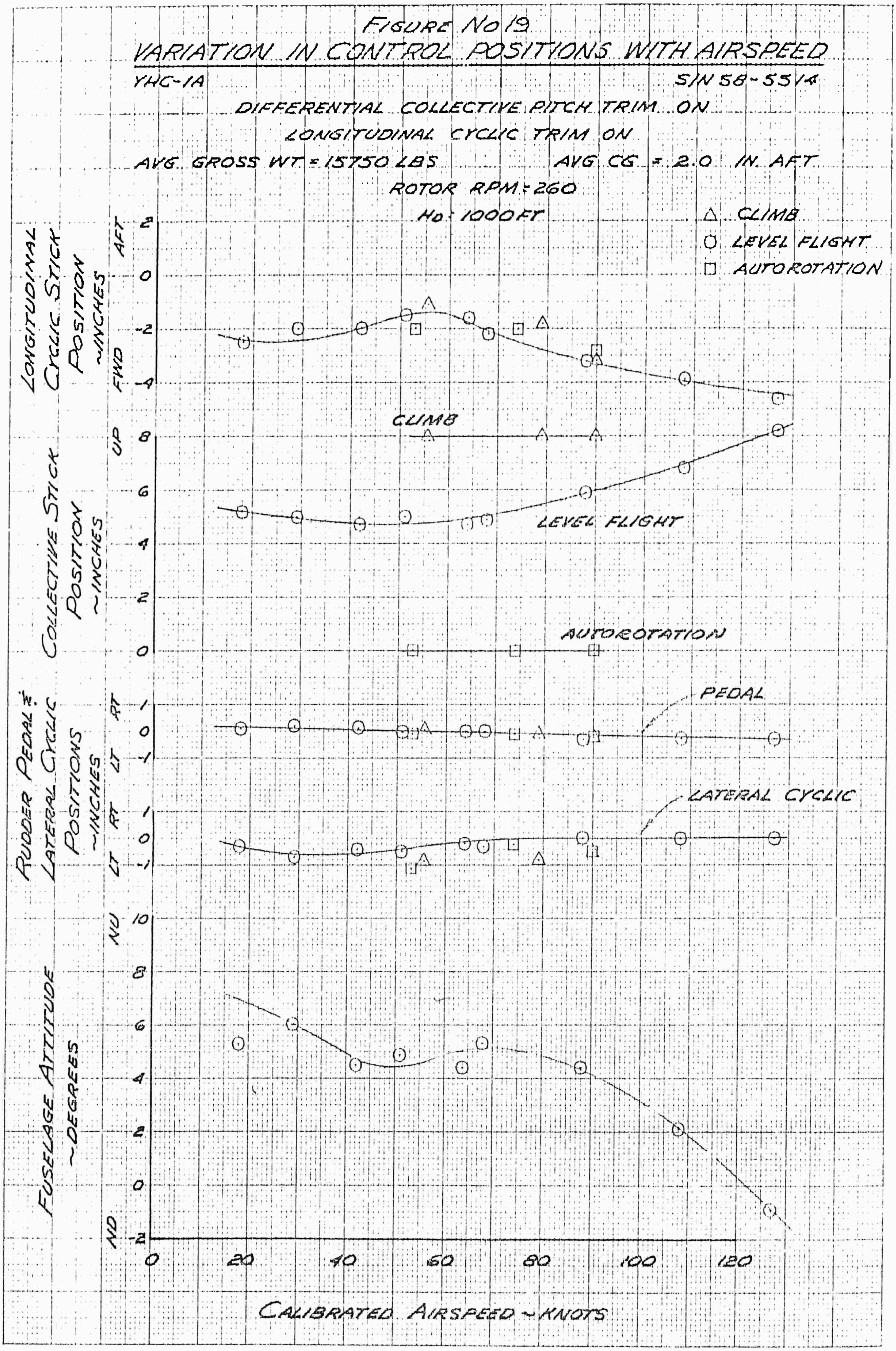


FIGURE No 19
VARIATION IN CONTROL POSITIONS WITH AIRSPEED
YHC-1A
S/N 58-5514
DIFFERENTIAL COLLECTIVE PITCH TRIM ON
LONGITUDINAL CYCLIC TRIM ON
AVG. GROSS WT = 15750 LBS
AVG CG = 2.0 IN. AFT
ROTOR RPM = 260
Ho: 1000 FT
△ CLIMB
○ LEVEL FLIGHT
□ AUTOROTATION
LONGITUDINAL CYCLIC STICK POSITION ~INCHES
AFT
FWD
CLIMB
LEVEL FLIGHT
AUTOROTATION
COLLECTIVE STICK POSITION ~INCHES
UP
PEDAL
LATERAL CYCLIC
RUDDER PEDAL & LATERAL CYCLIC POSITIONS ~INCHES
RT
LT
FUSELAGE ATTITUDE ~DEGREES
NU
ND
CALIBRATED AIRSPEED ~ KNOTS

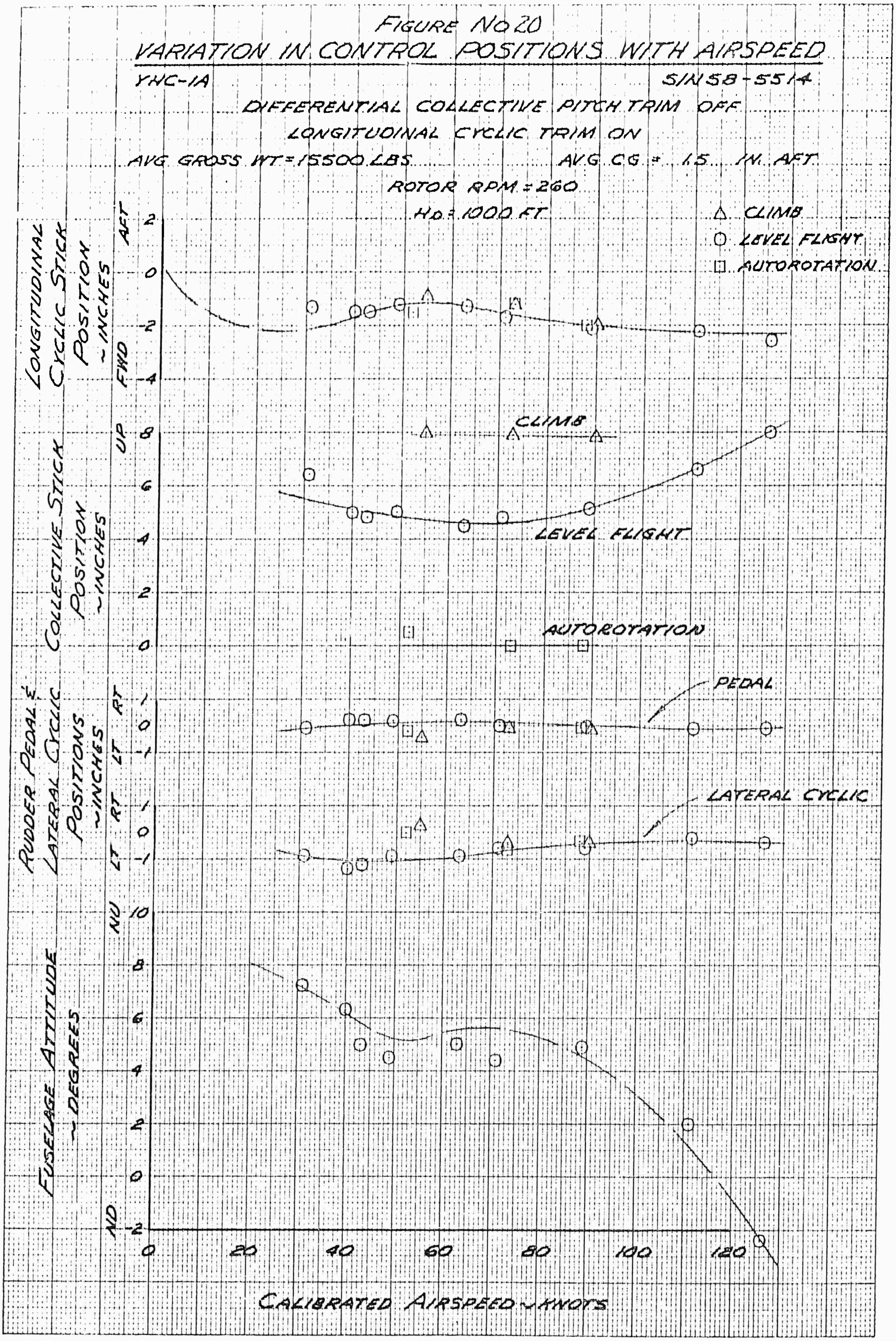


FIGURE NO 20
VARIATION IN CONTROL POSITIONS WITH AIRSPEED
YHC-1A
S/N 58-5514
DIFFERENTIAL COLLECTIVE PITCH TRIM OFF
LONGITUDINAL CYCLIC TRIM ON
AVG GROSS WT = 15500 LBS
AVG CG = 1.5 IN. AFT
ROTOR RPM = 260
Hp = 1000 FT
CLIMB
LEVEL FLIGHT
AUTOROTATION
LONGITUDINAL CYCLIC STICK POSITION ~ INCHES
AFT
FWD
COLLECTIVE STICK POSITION ~ INCHES
UP
CLIMB
LEVEL FLIGHT
AUTOROTATION
RUDDER PEDAL & LATERAL CYCLIC POSITIONS ~ INCHES
RT
LT
PEDAL
LATERAL CYCLIC
FUSELAGE ATTITUDE ~ DEGREES
NU
ND
0
20
40
60
80
100
120
CALIBRATED AIRSPEED ~ KNOTS

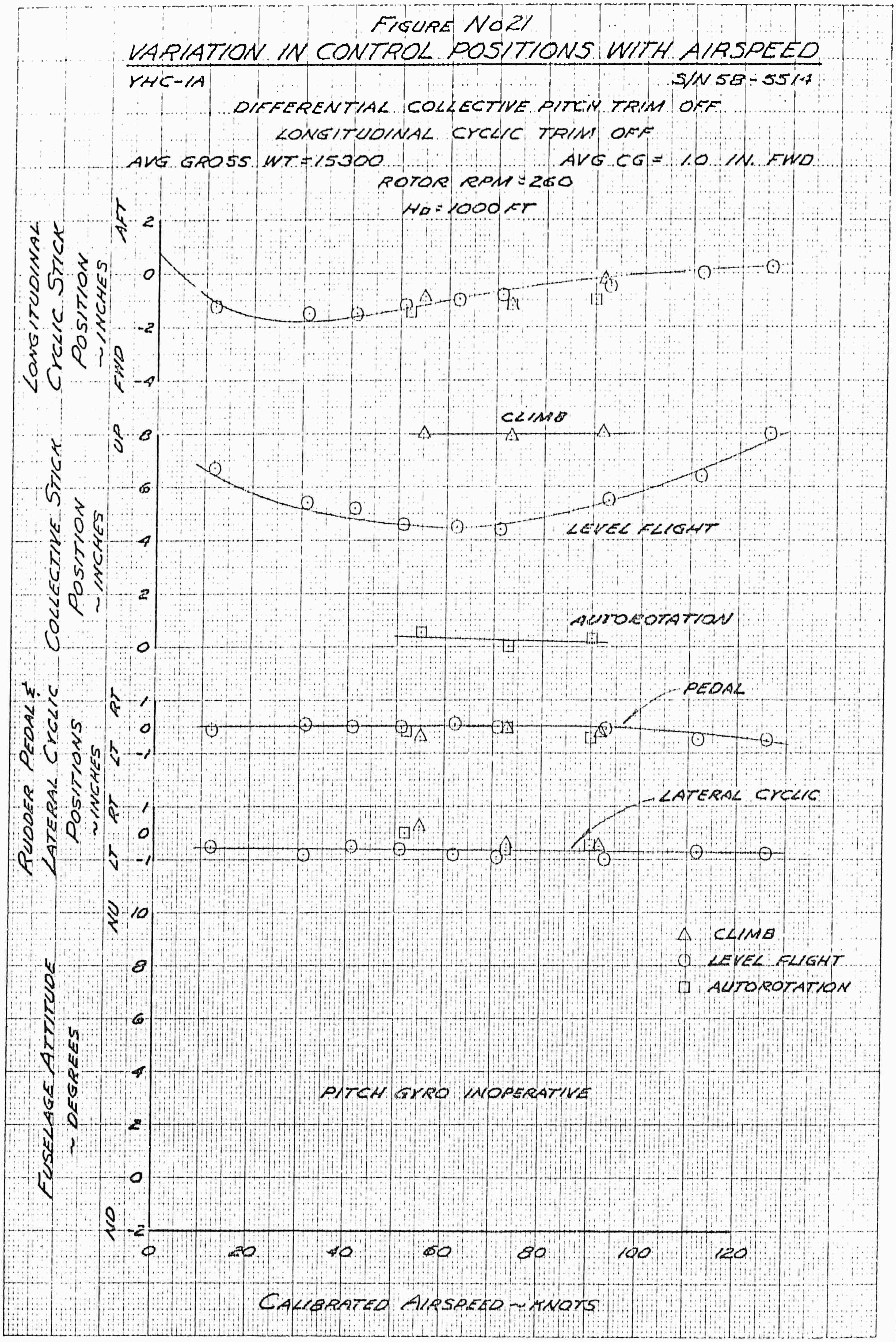
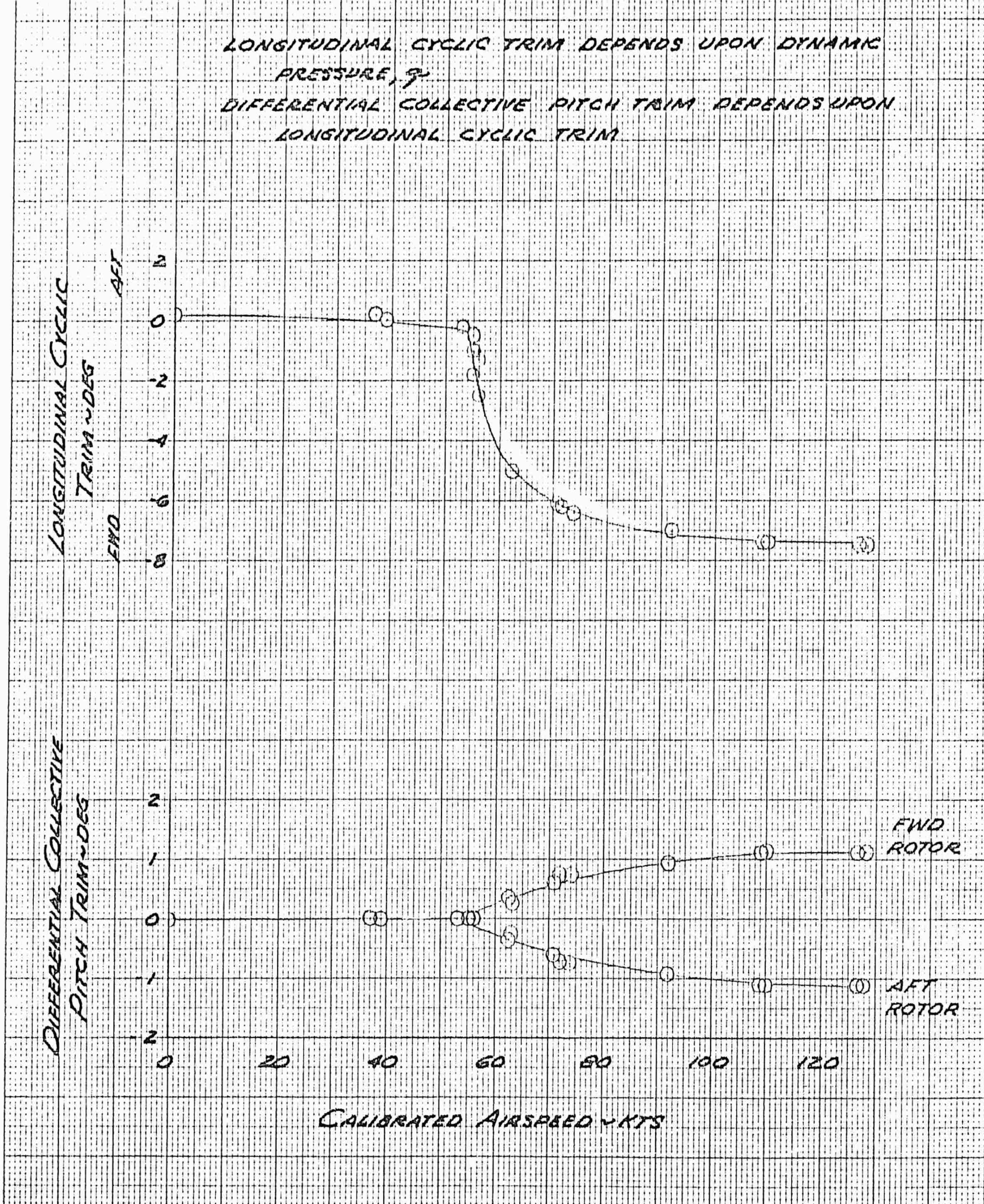


FIGURE No21
VARIATION IN CONTROL POSITIONS WITH AIRSPEED
YHC-1A
S/N 58-5514
DIFFERENTIAL COLLECTIVE PITCH TRIM OFF
LONGITUDINAL CYCLIC TRIM OFF
AVG GROSS WT = 15300
AVG CG = 1.0 IN. FWD
ROTOR RPM = 260
Hp = 1000 FT
LONGITUDINAL CYCLIC STICK POSITION ~ INCHES
AFT
FWD
COLLECTIVE STICK POSITION ~ INCHES
UP
CLIMB
LEVEL FLIGHT
AUTOROTATION
RUDDER PEDAL & LATERAL CYCLIC POSITIONS ~ INCHES
RT
LT
PEDAL
LATERAL CYCLIC
FUSELAGE ATTITUDE ~ DEGREES
NU
ND
PITCH GYRO INOPERATIVE
CLIMB
LEVEL FLIGHT
AUTOROTATION
CALIBRATED AIRSPEED ~ KNOTS
0
20
40
60
80
100
120

FIGURE No 22

LONGITUDINAL CYCLIC TRIM AND DIFFERENTIAL COLLECTIVE PITCH TRIM

YHC-1A S/N 58-5514



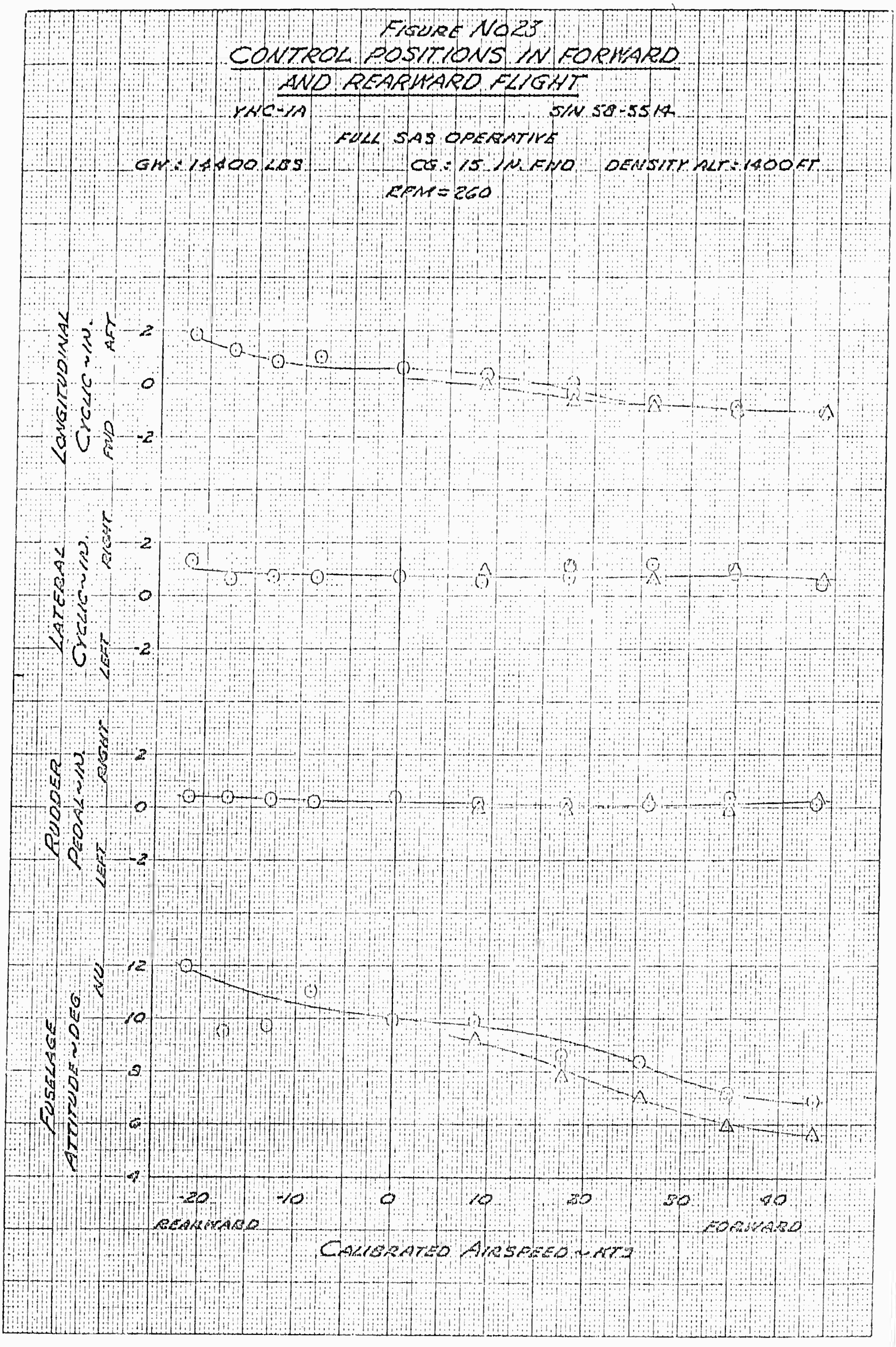


FIGURE NO 23
CONTROL POSITIONS IN FORWARD AND REARWARD FLIGHT
YHC-1A
S/N 58-5514
FULL SAS OPERATIVE
GW = 14400 LBS
CG = 15 IN FWD
DENSITY ALT = 1400 FT
RPM = 260
LONGITUDINAL CYCLIC ~ IN.
AFT
FWD
LATERAL CYCLIC ~ IN.
RIGHT
LEFT
RUDDER PEDAL ~ IN.
RIGHT
LEFT
FUSELAGE ATTITUDE ~ DEG
NU
-20
-10
0
10
20
30
40
REARWARD
FORWARD
CALIBRATED AIRSPEED ~ KTS

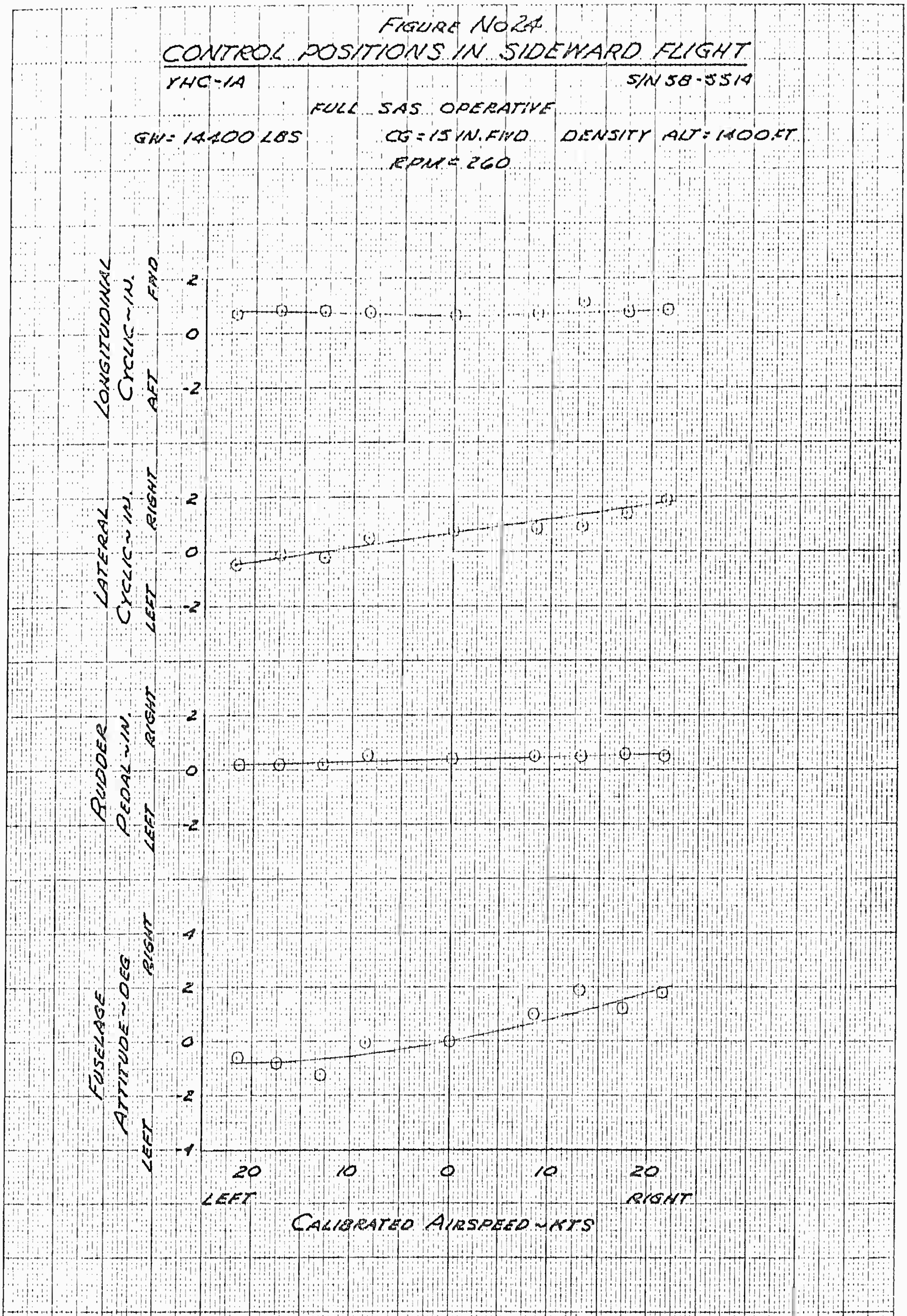


FIGURE No 24
CONTROL POSITIONS IN SIDEWARD FLIGHT
YHC-1A
S/N 58-5514
FULL SAS OPERATIVE
GW = 14400 LBS
CG = 15 IN. FWD
DENSITY ALT = 1400 FT
RPM = 260
LONGITUDINAL CYCLIC ~ IN.
AFT
FWD
LATERAL CYCLIC ~ IN.
LEFT
RIGHT
RUDDER PEDAL ~ IN.
LEFT
RIGHT
FUSELAGE ATTITUDE ~ DEG
LEFT
RIGHT
20
10
0
10
20
LEFT
RIGHT
CALIBRATED AIRSPEED ~ KTS

FIGURE NO 25

SUMMARY OF DIRECTIONAL STABILITY

YHC-1A S/N 58-5514

FULL SAS OPERATING

DATA OBTAINED FROM FIGURES 27 THROUGH 31

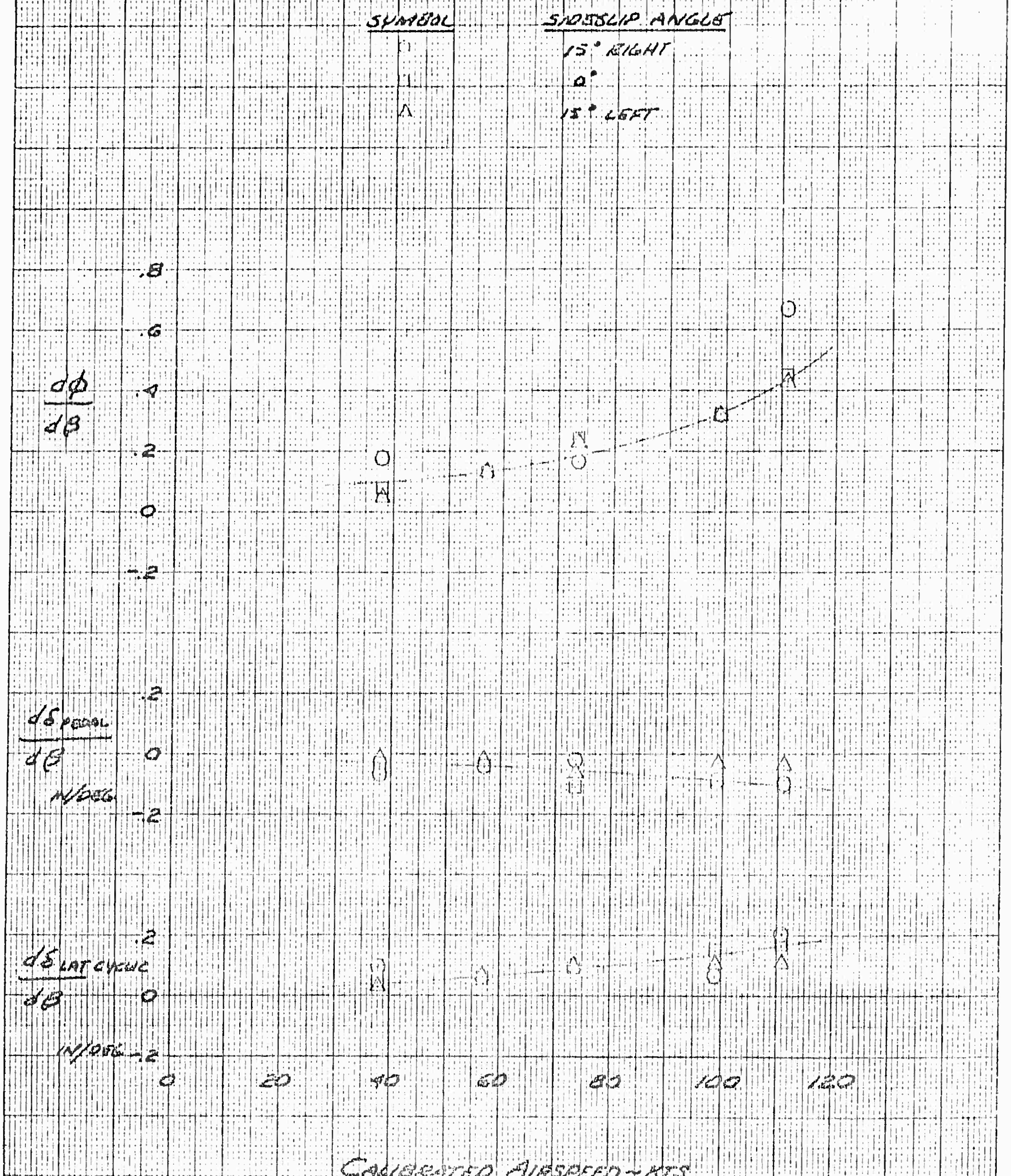


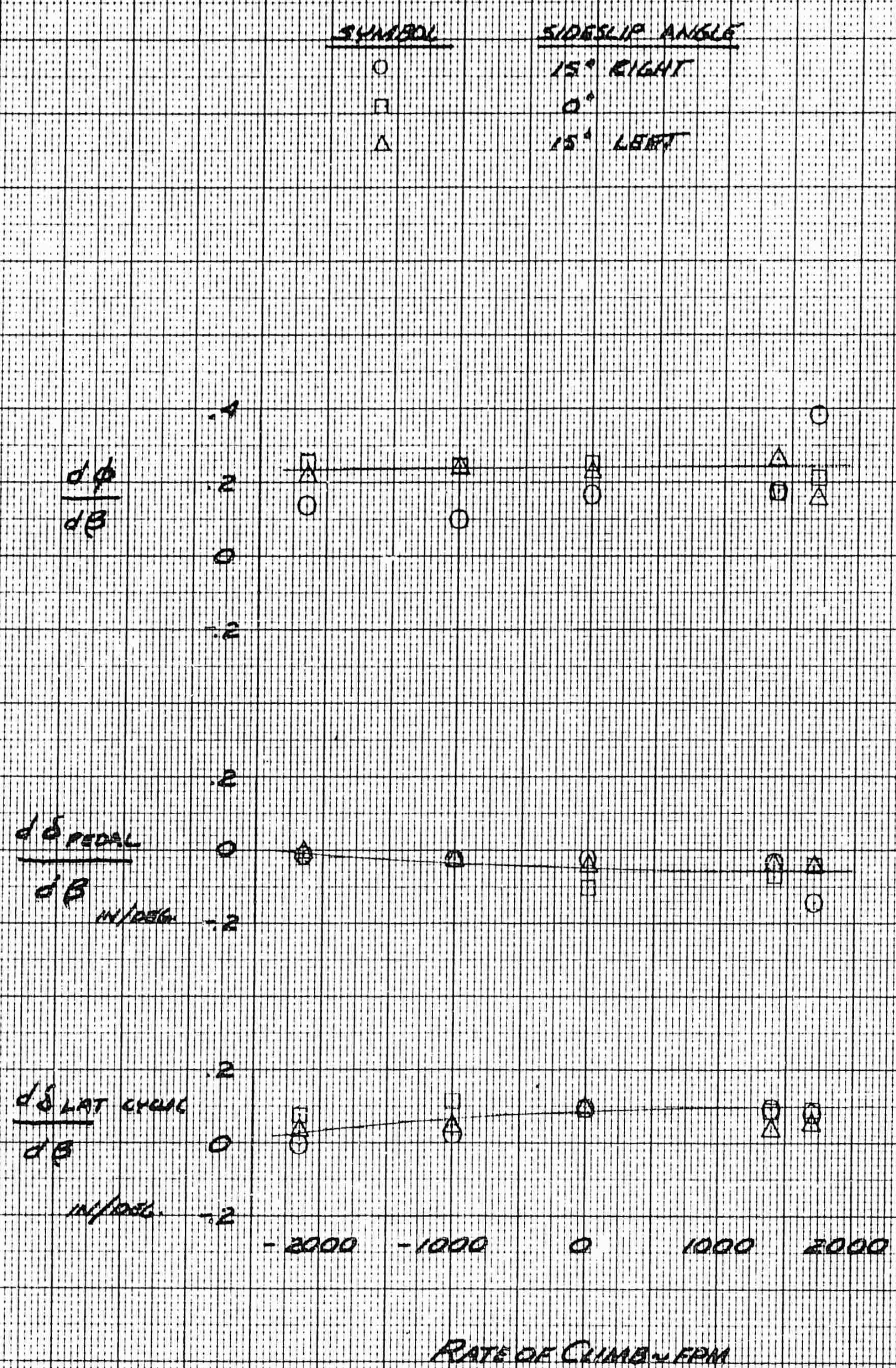
FIGURE NO 26

SUMMARY OF DIRECTIONAL STABILITY

YHC-1A S/N 58-5514

FULL SAS OPERATING

DATA OBTAINED FROM FIGURES 32 THROUGH 36



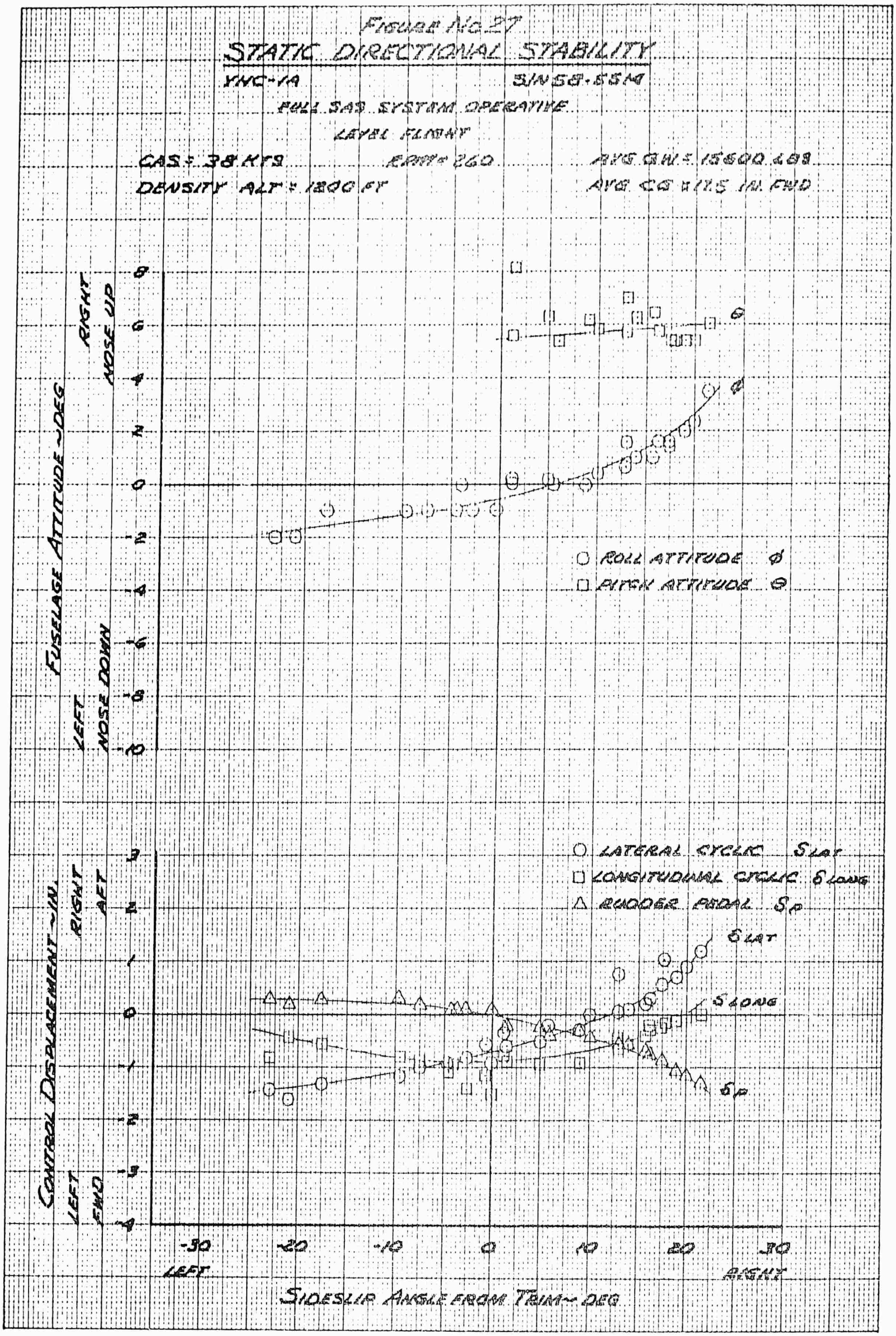


FIGURE No. 27
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
LEVEL FLIGHT
CAS = 38 KTS
RPM = 260
AVG G.W. = 15600 LBS
DENSITY ALT = 1800 FT
AVG CG = 11.5 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE ϕ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δP
SIDESLIP ANGLE FROM TRIM ~ DEG
LEFT
RIGHT

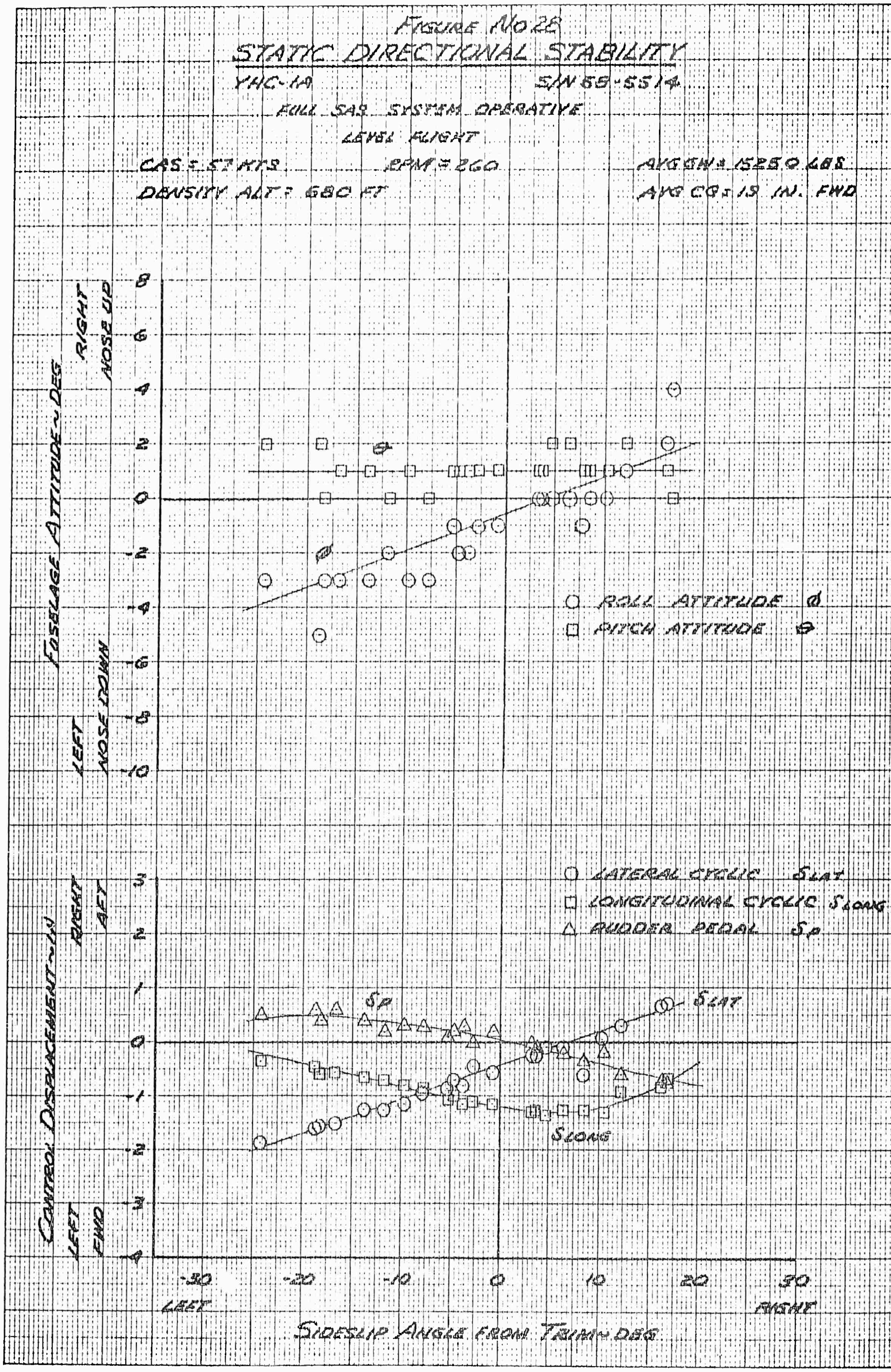


FIGURE NO 28
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 55-5514
FULL SAS SYSTEM OPERATIVE
LEVEL FLIGHT
CAS = 57 KTS
RPM = 260
AVG GW = 15250 LBS
DENSITY ALT = 680 FT
AVG CG = 13 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN
RIGHT
AFT
LEFT
FWD
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δP
δP
δLAT
δLONG
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

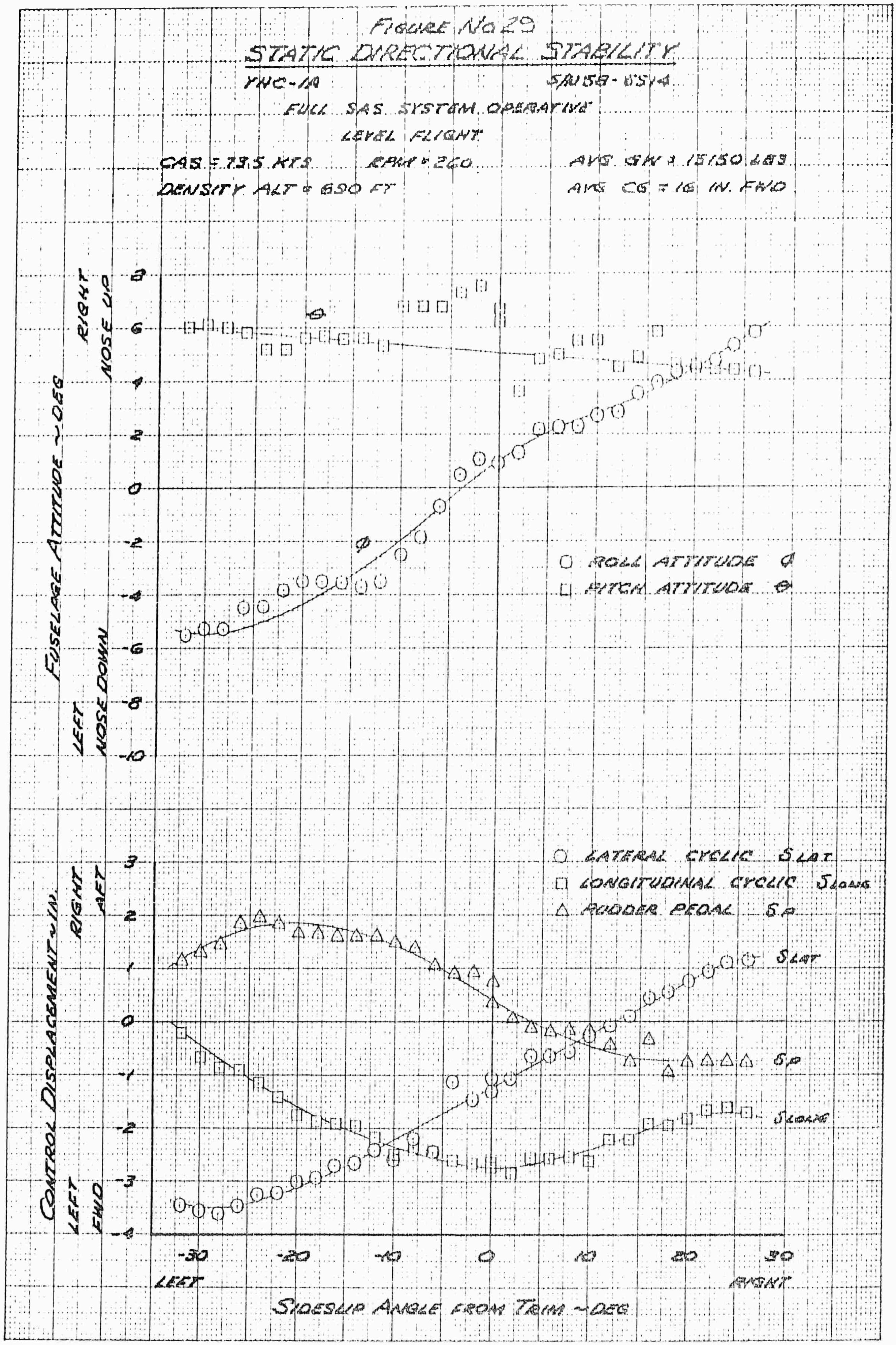


FIGURE No 29
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
LEVEL FLIGHT
CAS = 73.5 KTS
RPM = 260
AVG GW = 15150 LBS
DENSITY ALT = 650 FT
AVG CG = 16 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δP
SIDESLIP ANGLE FROM TRIM ~ DEG
LEFT
RIGHT

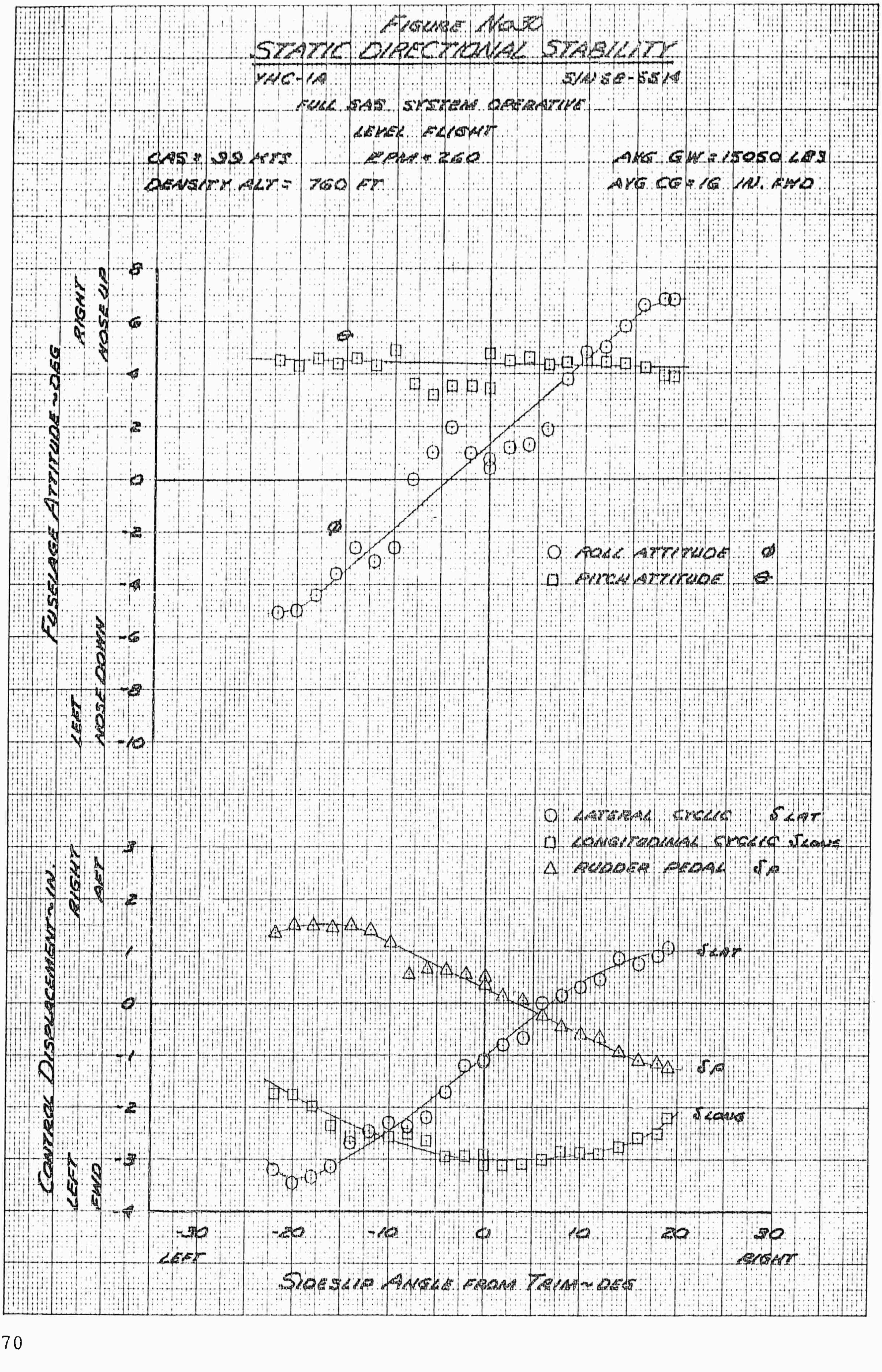


FIGURE No.30
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
LEVEL FLIGHT
CAS = 99 KTS
RPM = 260
AVG GW = 15050 LBS
DENSITY ALT = 760 FT
AVG CG = 16 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
8
6
4
2
0
-2
-4
-6
-8
-10
φ
θ
○ ROLL ATTITUDE φ
□ PITCH ATTITUDE θ
○ LATERAL CYCLIC δLAT
□ LONGITUDINAL CYCLIC δLONG
△ RUDDER PEDAL δP
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
3
2
1
0
-1
-2
-3
-4
δLAT
δP
δLONG
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

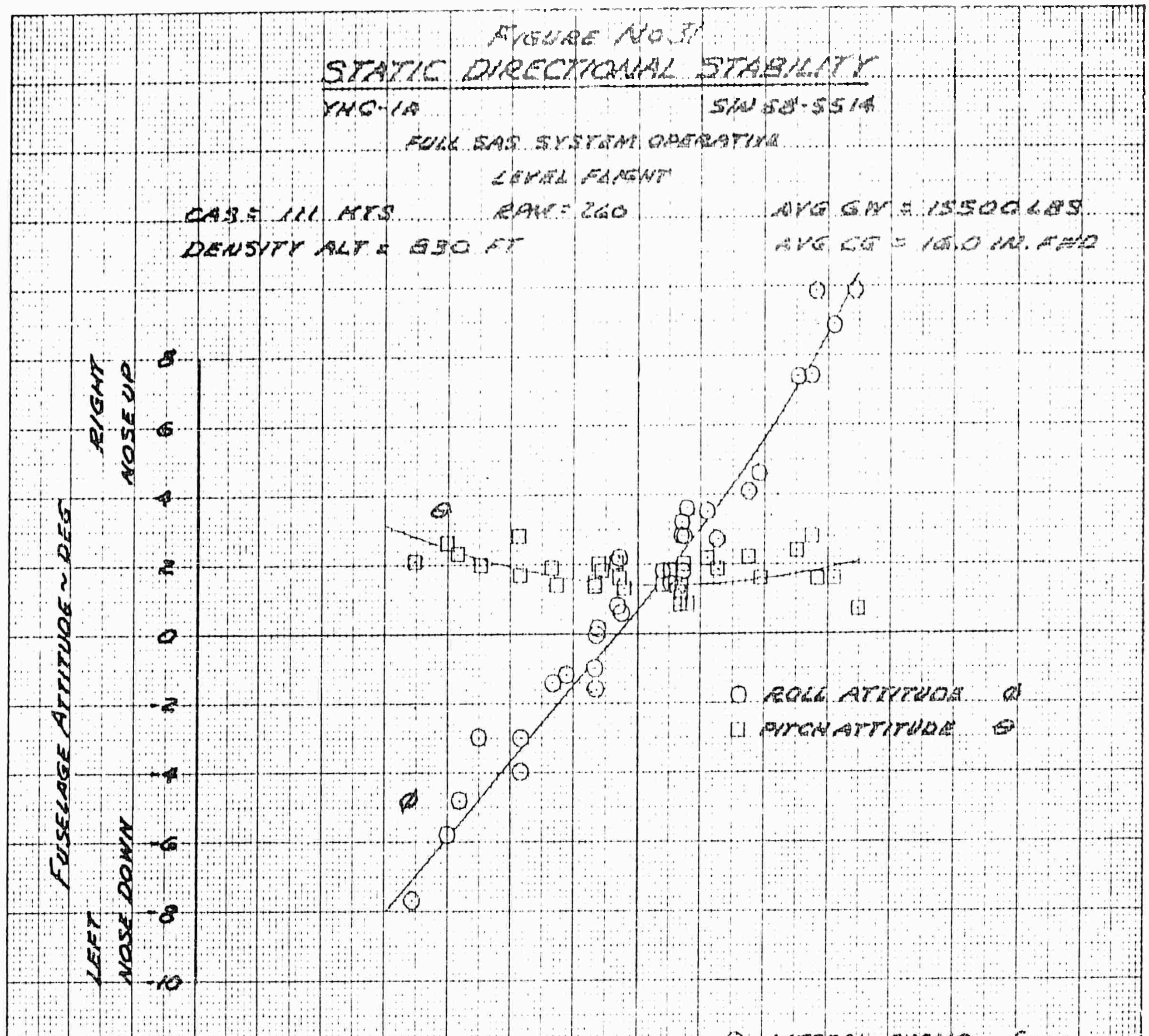


FIGURE NO 31
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATING
LEVEL FLIGHT
CAS = 111 KTS
RPM = 260
AVG GW = 15500 LBS
DENSITY ALT = 530 FT
AVG CG = 16.0 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT NOSE UP
LEFT NOSE DOWN
8
6
4
2
0
-2
-4
-6
-8
-10
○ ROLL ATTITUDE Φ
□ PITCH ATTITUDE Θ
○ LATERAL CYCLIC δLAT
□ LONGITUDINAL CYCLIC δLONG
△ RUDDER PEDAL δP
CONTROL DISPLACEMENT ~ IN
RIGHT AFT
LEFT FWD
3
2
1
0
-1
-2
-3
-4
δLAT
δP
δLONG
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG.

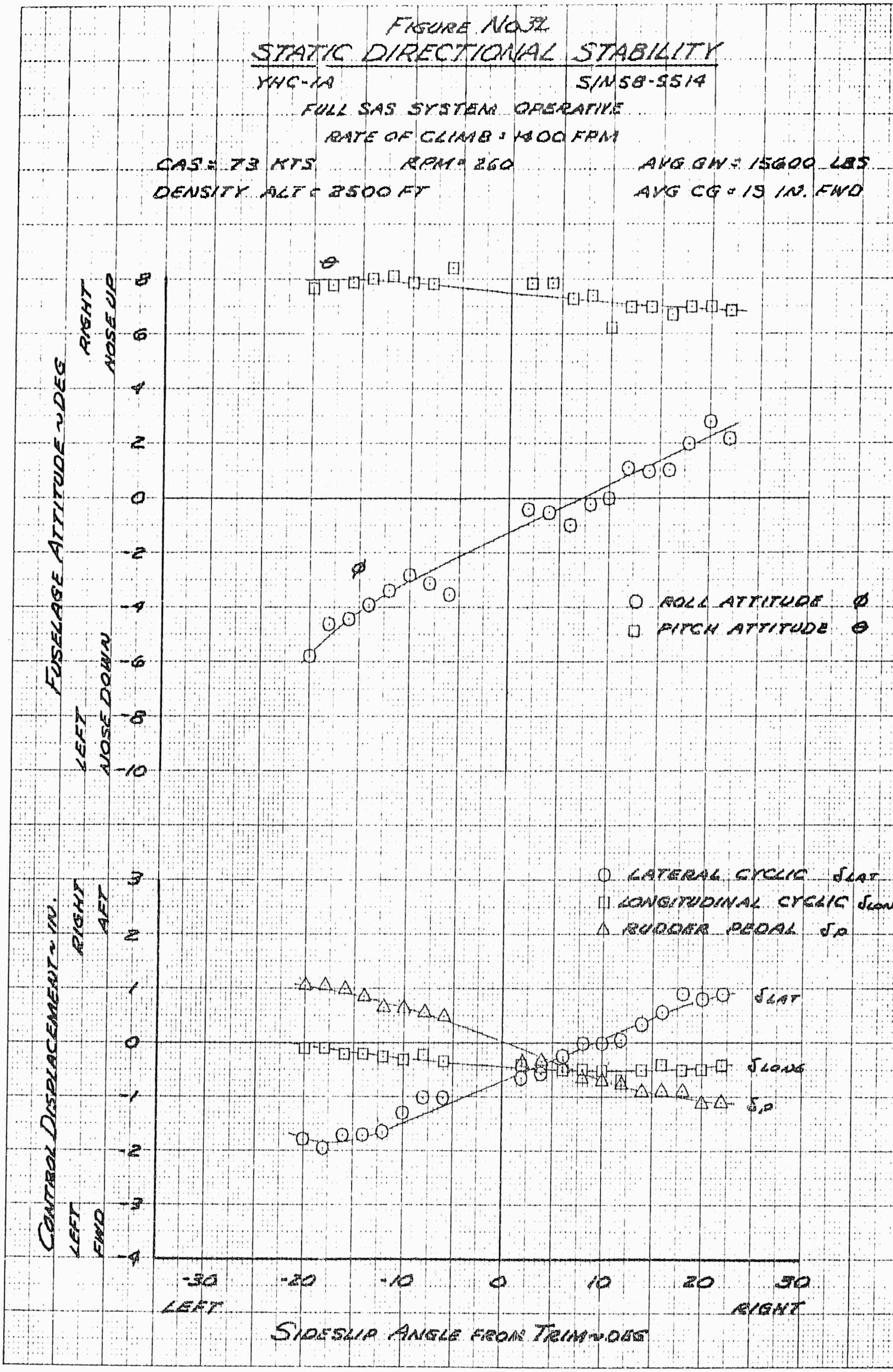
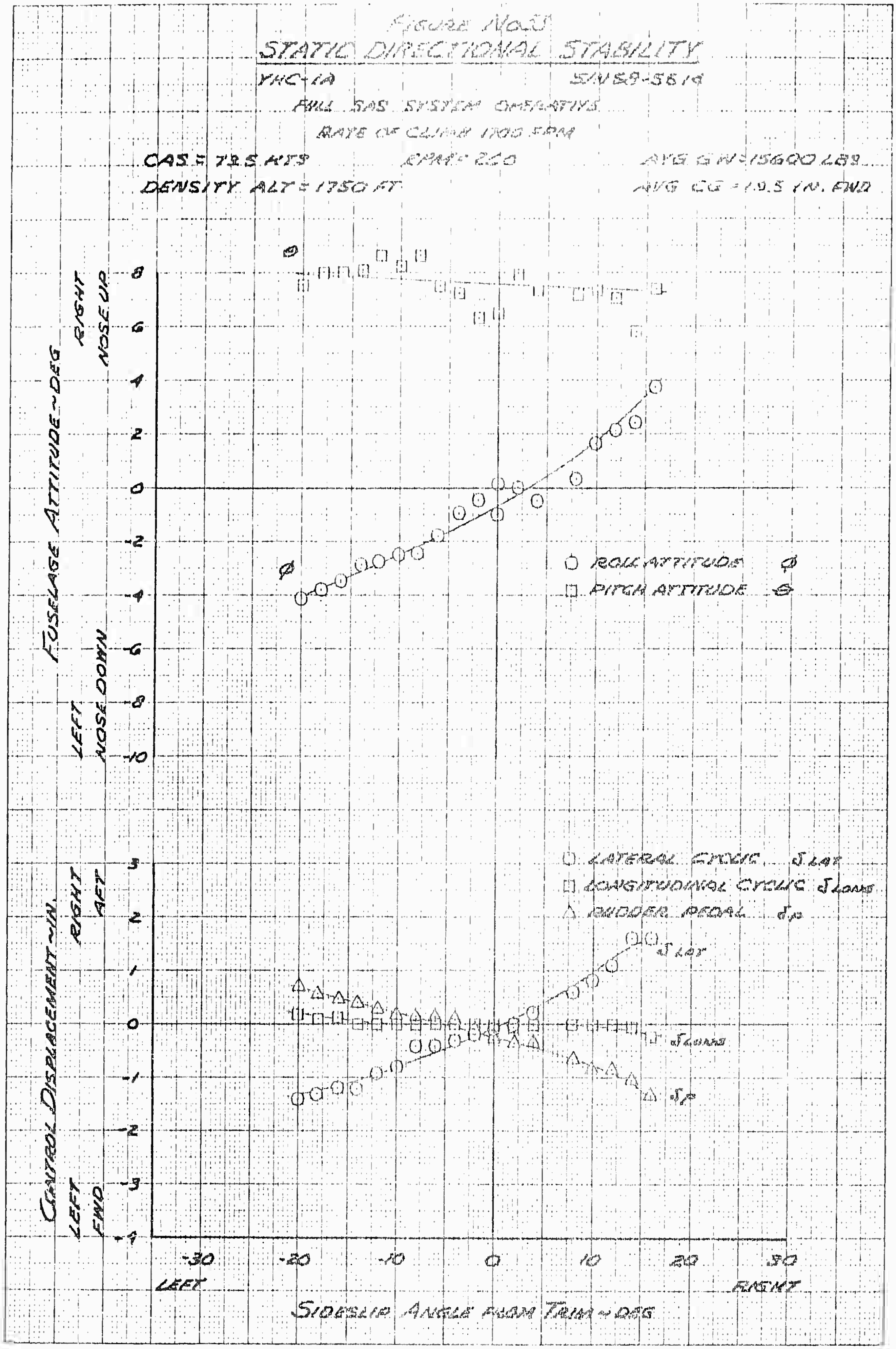


FIGURE No. 32
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
RATE OF CLIMB = 1400 FPM
CAS = 73 KTS
RPM = 260
AVG GW = 15600 LBS
DENSITY ALT = 2500 FT
AVG CG = 13 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δP
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG



STATIC DIRECTIONAL STABILITY
YHC-1A
CAS = 72.5 KTS
DENSITY ALT = 1750 FT
AVG GW = 15600 LBS
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δp
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

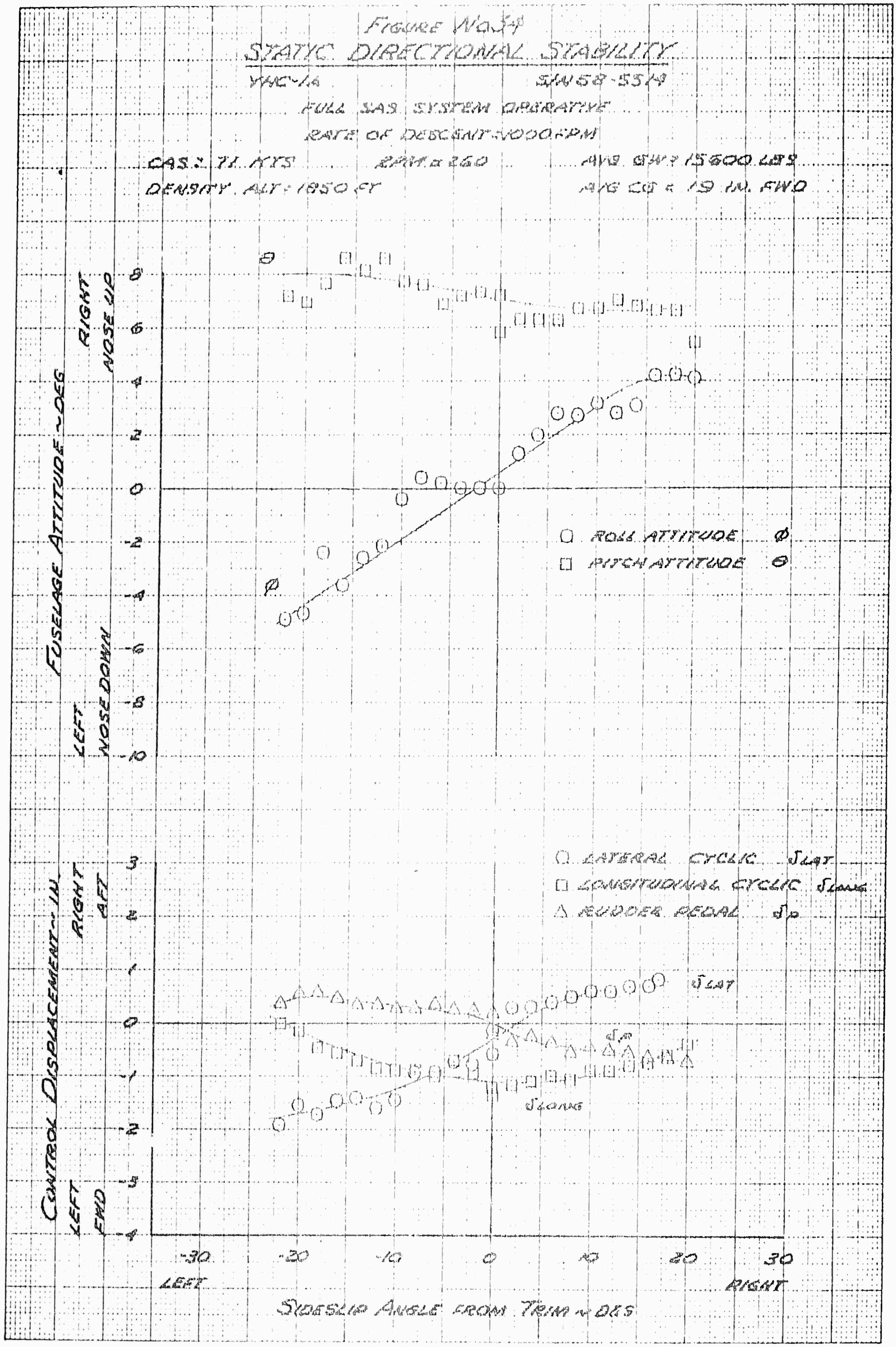
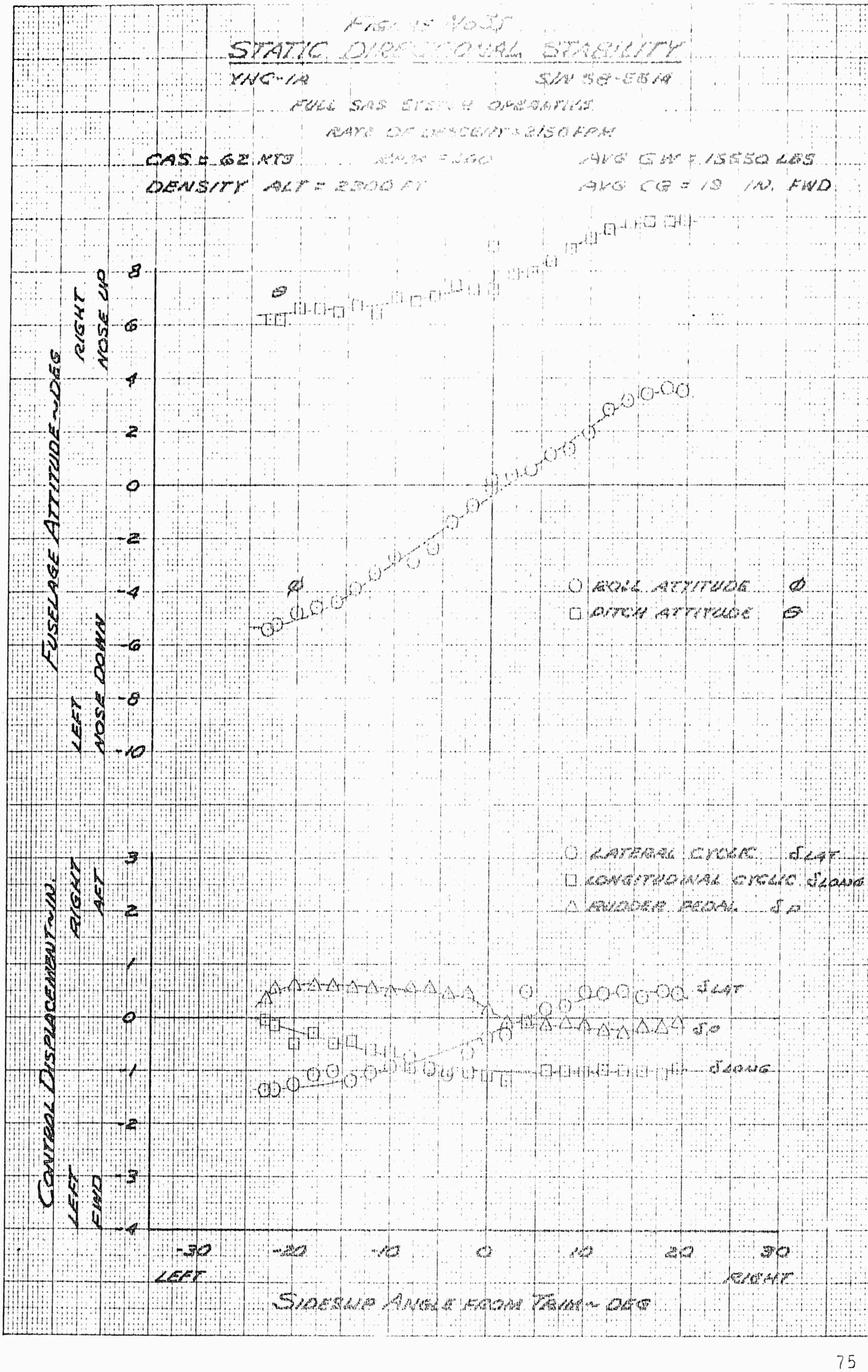


FIGURE NO. 34
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
RATE OF DESCENT ≈ 1000 FPM
CAS ≈ 71 KTS
RPM ≈ 260
AVG GW ≈ 15600 LBS
DENSITY ALT ≈ 1850 FT
AVG CG ≈ 19 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
8
6
4
2
0
-2
-4
-6
-8
-10
θ
φ
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN
RIGHT
AFT
LEFT
FWD
3
2
1
0
-1
-2
-3
-4
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δP
δLAT
δP
δLONG
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG


STATIC DIRECTIONAL STABILITY
YHC-1A
CAS = 62 KTS
DENSITY ALT = 2300 FT
AVG GW = 15550 LBS
AVG CG = 13 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
LATERAL CYCLIC δLAT
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δP
δLAT
δP
δLONG
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

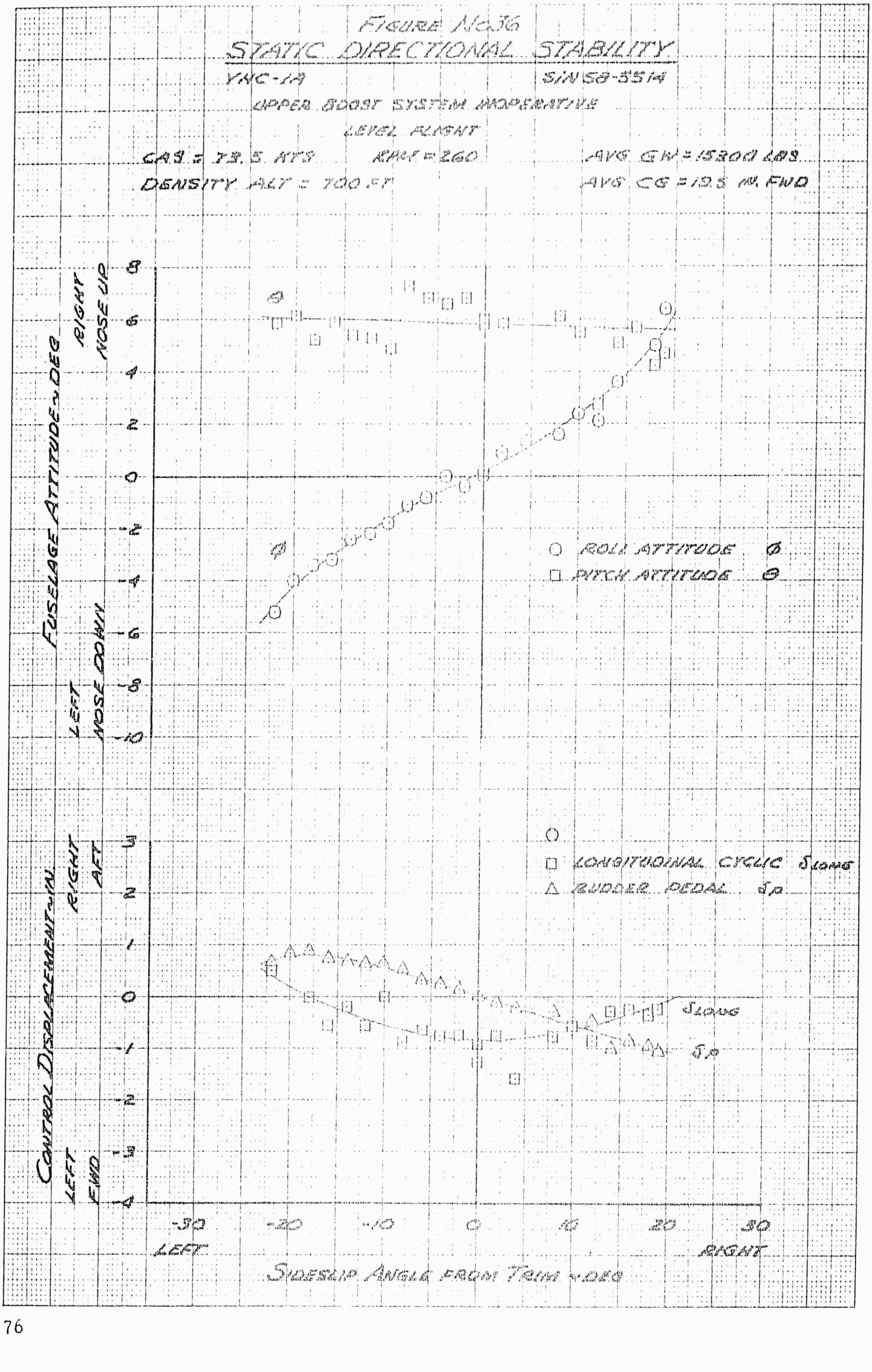


FIGURE No. 36
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
UPPER BOOST SYSTEM INOPERATIVE
LEVEL FLIGHT
CAS = 73.5 KTS
RPM = 260
AVG GW = 15300 LBS
DENSITY ALT = 700 FT
AVG CG = 19.5 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE ϕ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
LONGITUDINAL CYCLIC δ_{LONG}
RUDDER PEDAL δ_R
δ_{LONG}
δ_R
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

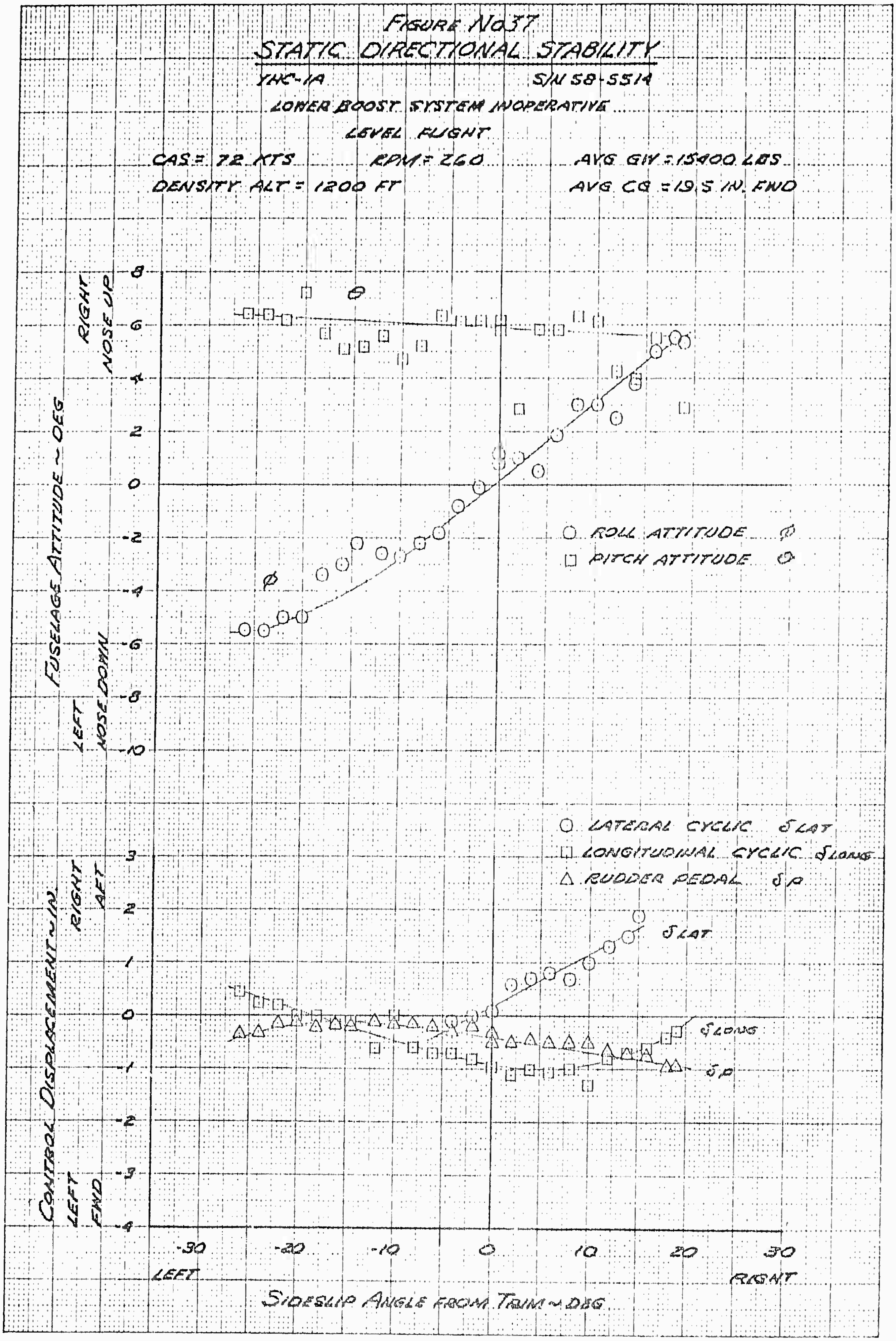


FIGURE No37
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
LOWER BOOST SYSTEM INOPERATIVE
LEVEL FLIGHT
CAS = 72 KTS
RPM = 260
AVG GW = 15400 LBS
DENSITY ALT = 1200 FT
AVG CG = 19.5 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
8
6
4
2
0
-2
-4
-6
-8
-10
θ
φ
○ ROLL ATTITUDE φ
□ PITCH ATTITUDE θ
○ LATERAL CYCLIC δLAT
□ LONGITUDINAL CYCLIC δLONG
△ RUDDER PEDAL δP
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
3
2
1
0
-1
-2
-3
-4
δLAT
δLONG
δP
-30
-20
-10
0
10
20
30
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

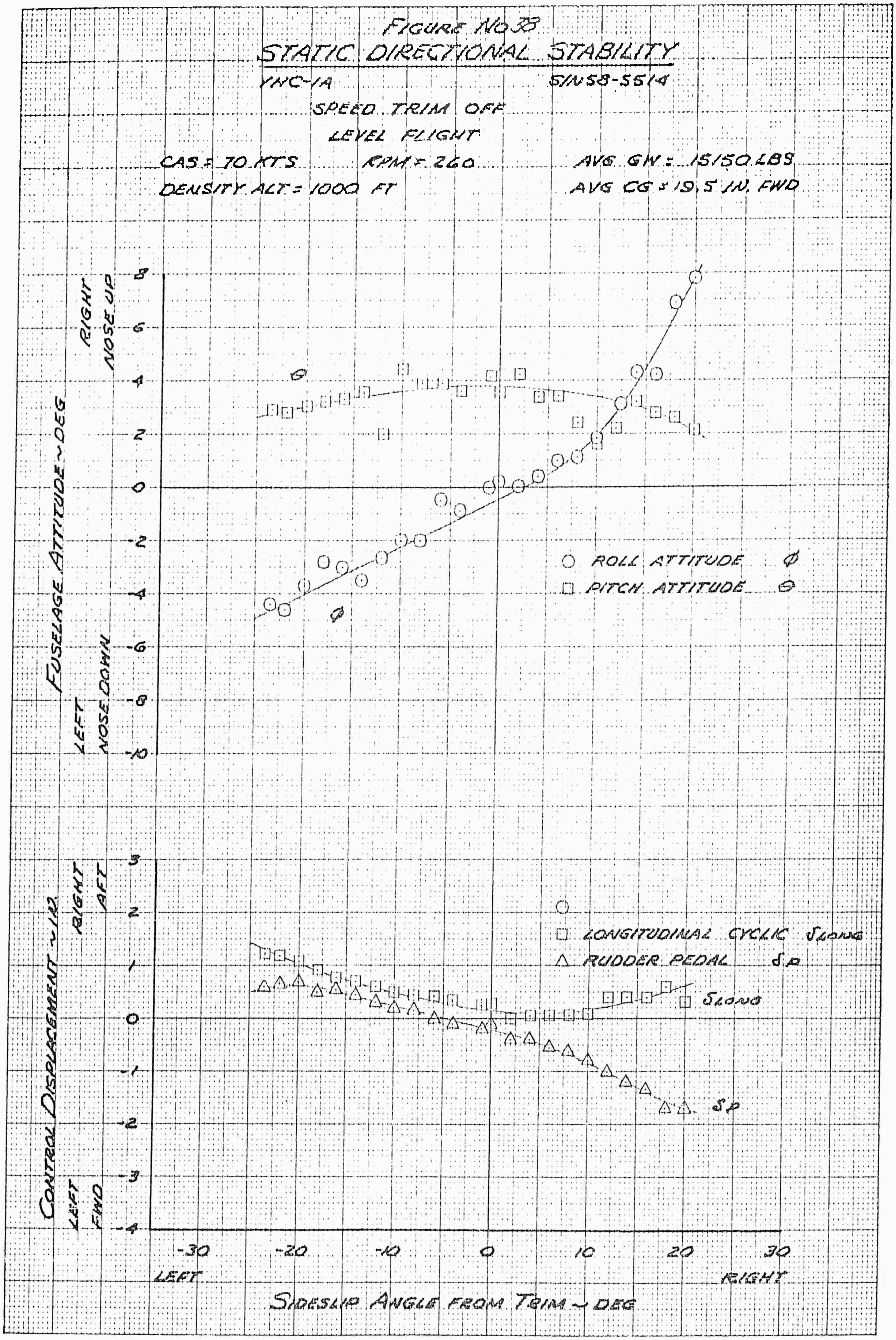


FIGURE NO 38
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
SPEED TRIM OFF
LEVEL FLIGHT
CAS = 70 KTS
RPM = 260
AVG GW = 15150 LBS
DENSITY ALT = 1000 FT
AVG CG = 19.5 IN. FWD
FUSELAGE ATTITUDE ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
ROLL ATTITUDE φ
PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN
RIGHT
AFT
LEFT
FWD
LONGITUDINAL CYCLIC δLONG
RUDDER PEDAL δp
SIDESLIP ANGLE FROM TRIM ~ DEG
LEFT
RIGHT

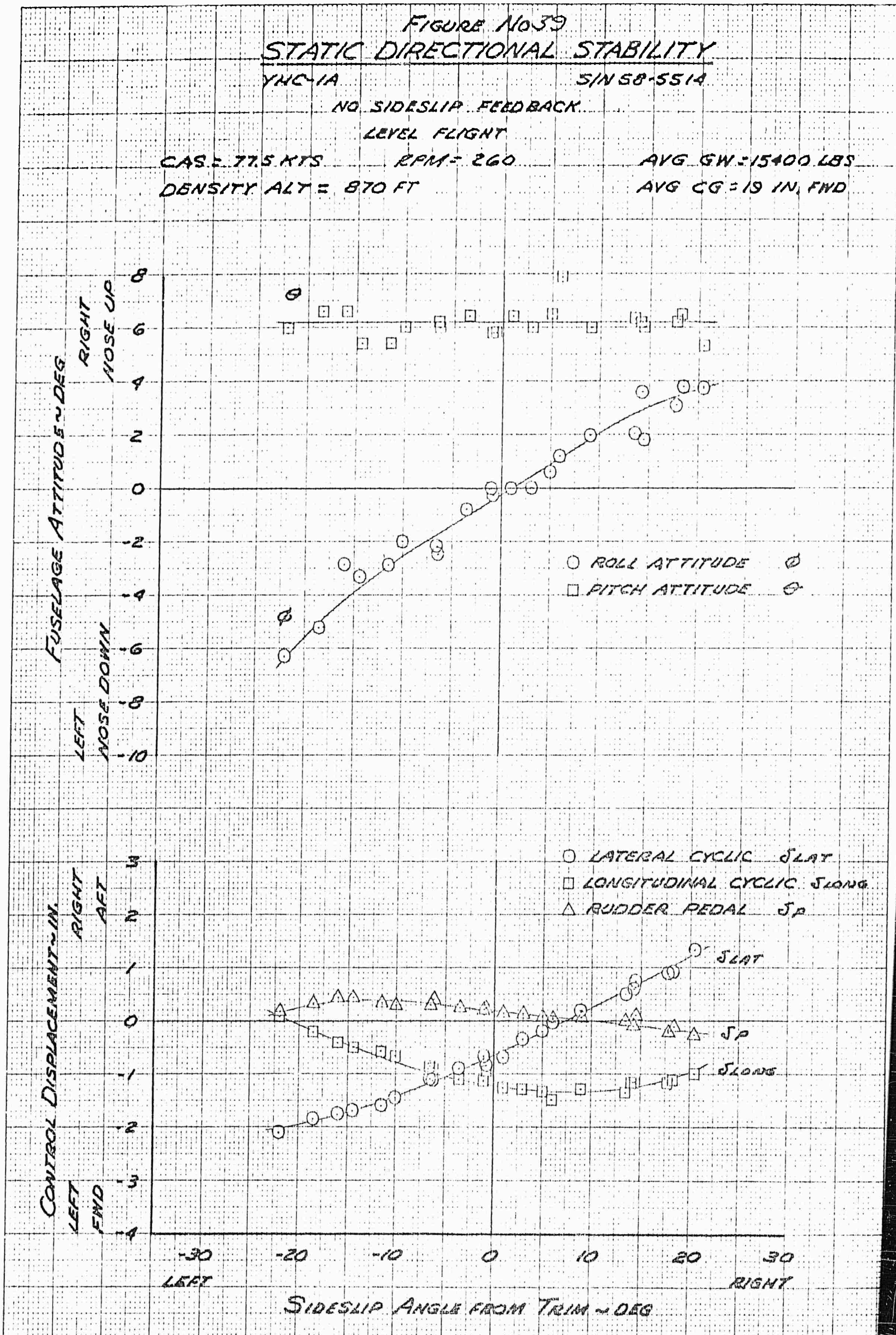


FIGURE No 39
STATIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
NO SIDESLIP FEEDBACK
LEVEL FLIGHT
CAS = 77.5 KTS
RPM = 260
AVG GW = 15400 LBS
DENSITY ALT = 870 FT
AVG CG = 19 IN FWD
FUSELAGE ATTITUDES ~ DEG
RIGHT
NOSE UP
LEFT
NOSE DOWN
○ ROLL ATTITUDE φ
□ PITCH ATTITUDE θ
CONTROL DISPLACEMENT ~ IN.
RIGHT
AFT
LEFT
FWD
○ LATERAL CYCLIC δLAT
□ LONGITUDINAL CYCLIC δLONG
△ RUDDER PEDAL δP
LEFT
RIGHT
SIDESLIP ANGLE FROM TRIM ~ DEG

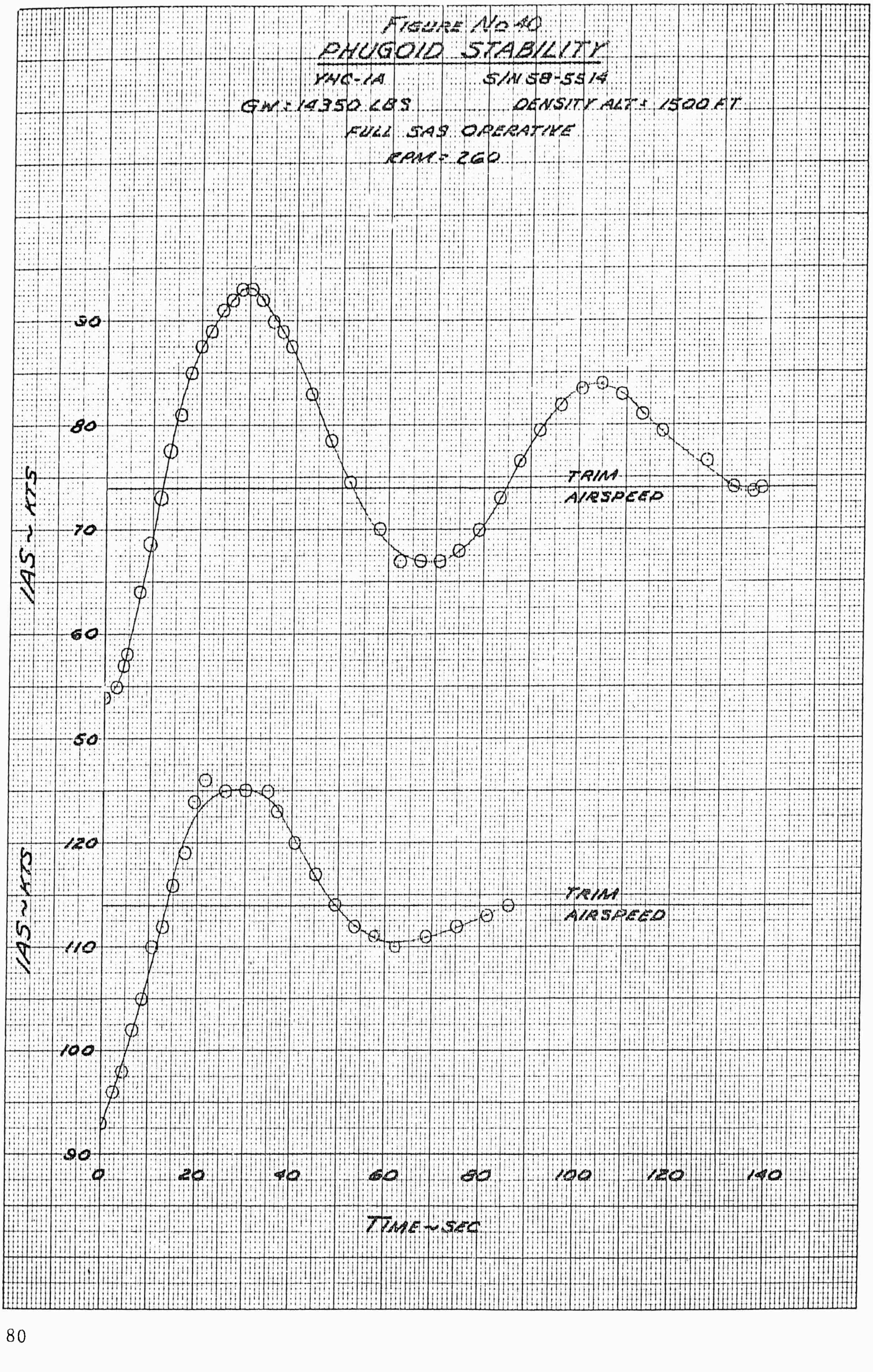


FIGURE No 40
PHUGOID STABILITY
YHC-1A
S/N 58-5514
GW: 14350 LBS
DENSITY ALT: 1500 FT
FULL SAS OPERATIVE
RPM = 260
IAS ~ KTS
TRIM AIRSPEED
TIME ~ SEC

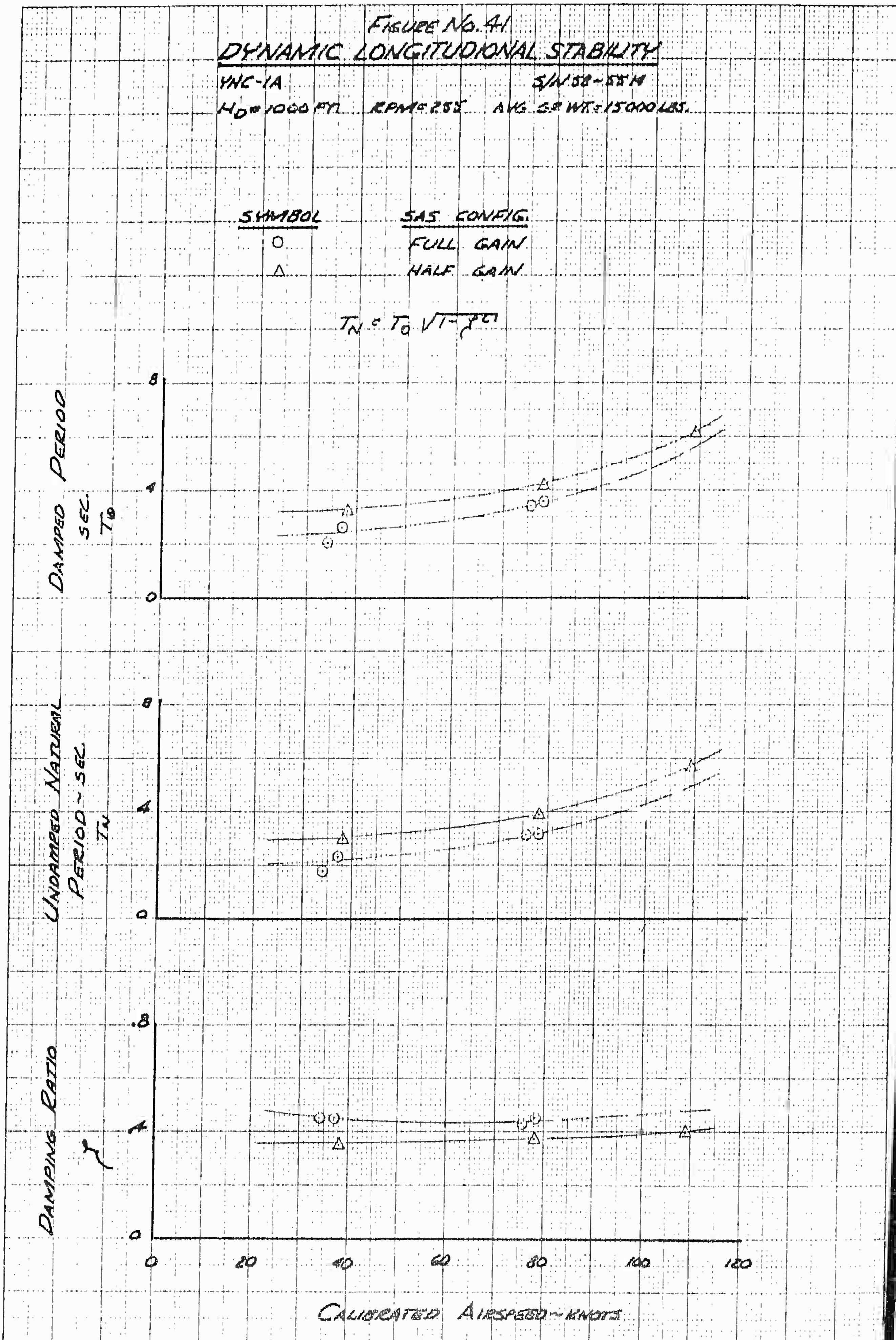


FIGURE NO. 41
DYNAMIC LONGITUDIONAL STABILITY
YHC-1A
S/N 58-5514
H_D= 1000 FT.
RPM=255
AVG. GR WT=15000 LBS.
SYMBOL
SAS CONFIG.
FULL GAIN
HALF GAIN
$T_N = T_0 \sqrt{1-\zeta^2}$
DAMPED PERIOD SEC. T_0
UNDAMPED NATURAL PERIOD ~ SEC. T_N
DAMPING RATIO ζ
CALIBRATED AIRSPEED ~ KNOTS

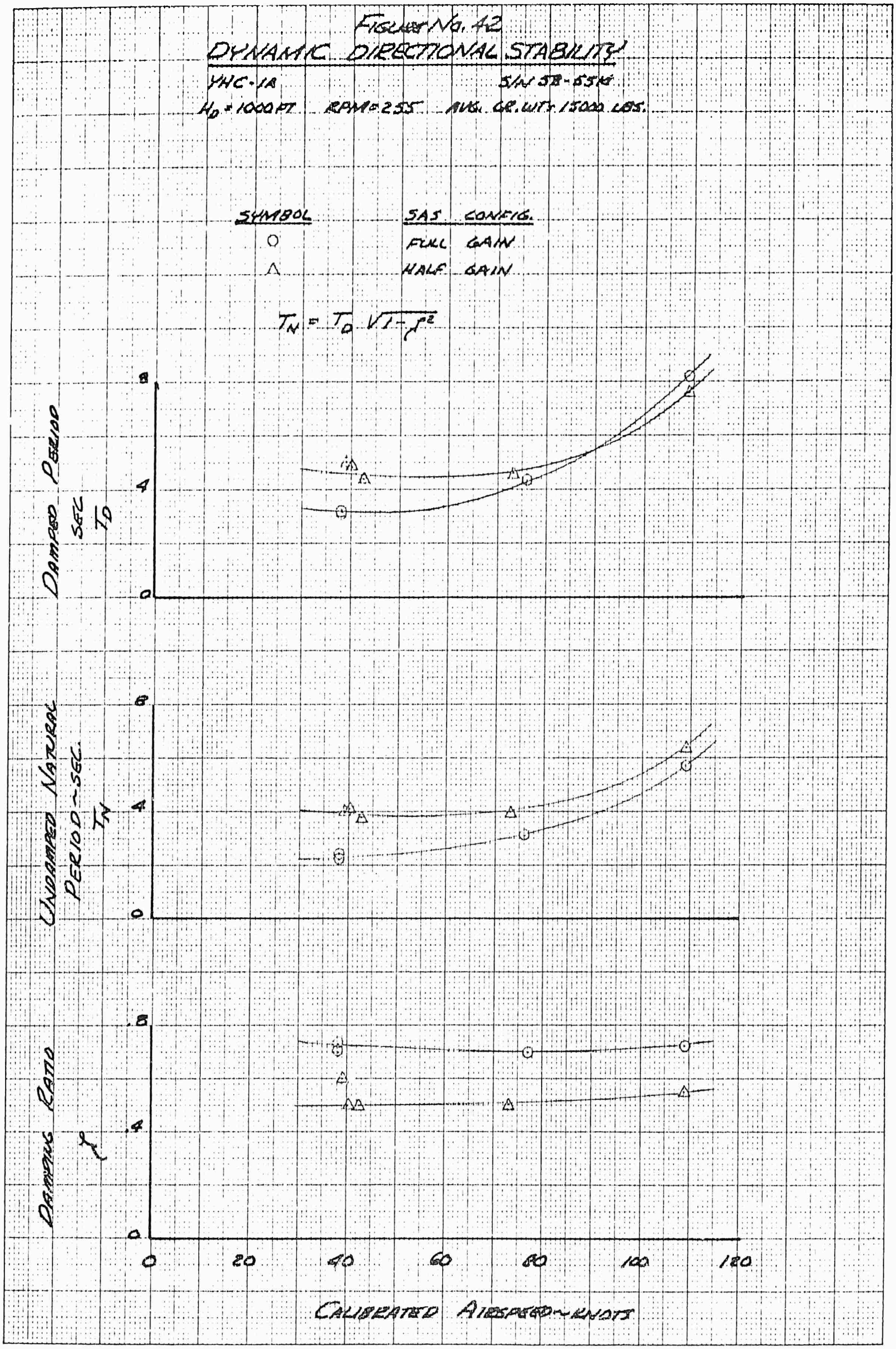


FIGURE No. 42
DYNAMIC DIRECTIONAL STABILITY
YHC-1A
S/N 58-5514
H_D = 1000 FT
RPM = 255
AVG. GR. WT. = 15000 LBS.
SYMBOL
SAS CONFIG.
FULL GAIN
HALF GAIN
$T_N = T_D \sqrt{1-\zeta^2}$
DAMPED PERIOD
SEC
T_D
UNDAMPED NATURAL PERIOD ~ SEC.
T_N
DAMPING RATIO
ζ
0
20
40
60
80
100
120
CALIBRATED AIRSPEED ~ KNOTS

Figure No. 43
LONGITUDINAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 75 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1900 Ft. Rotor RPM = 255
SAS Operating at Full-Gain

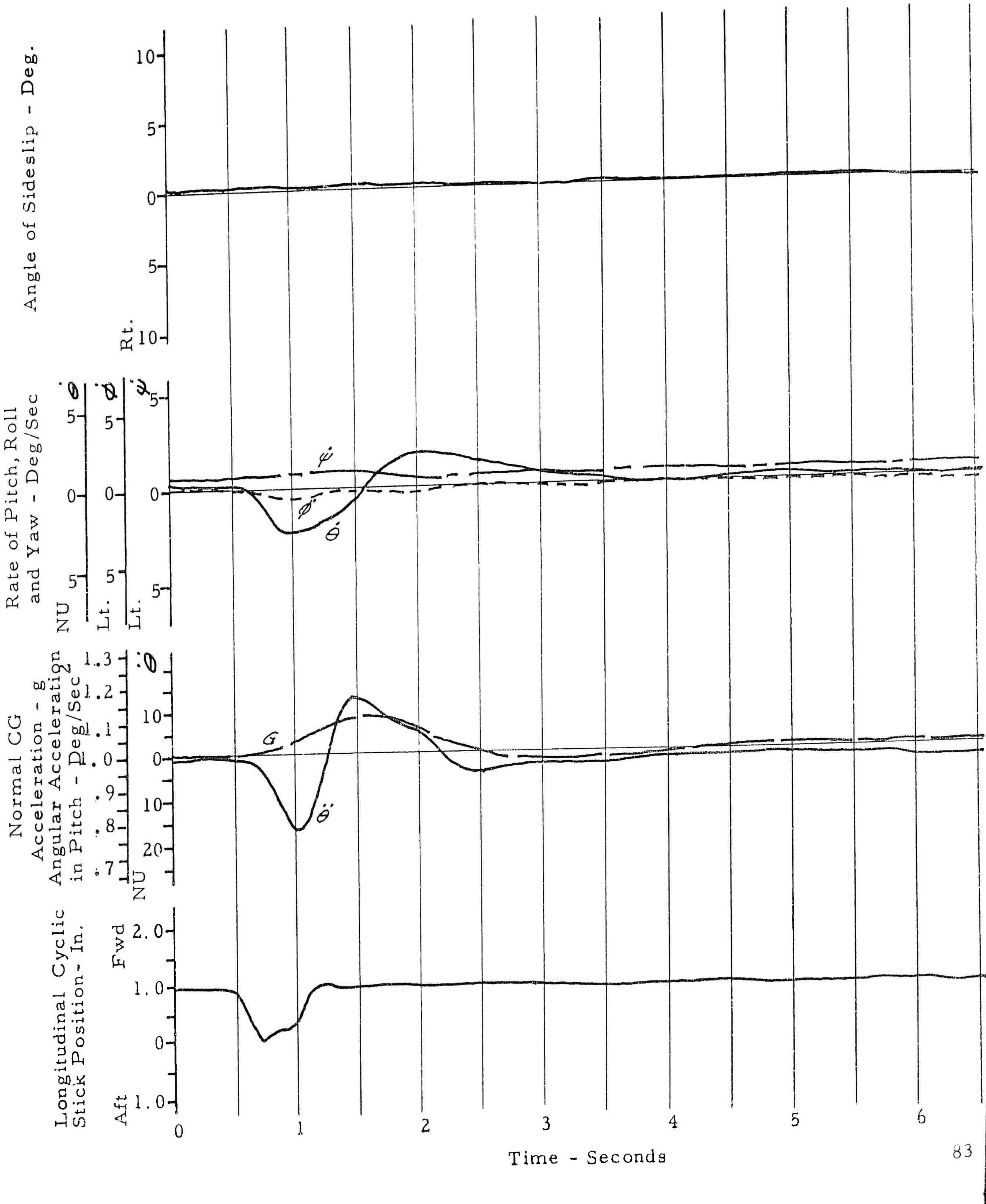


Figure No. 44
LONGITUDINAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 75 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1950 Ft. Rotor RPM = 255
SAS Operating at Full-Gain

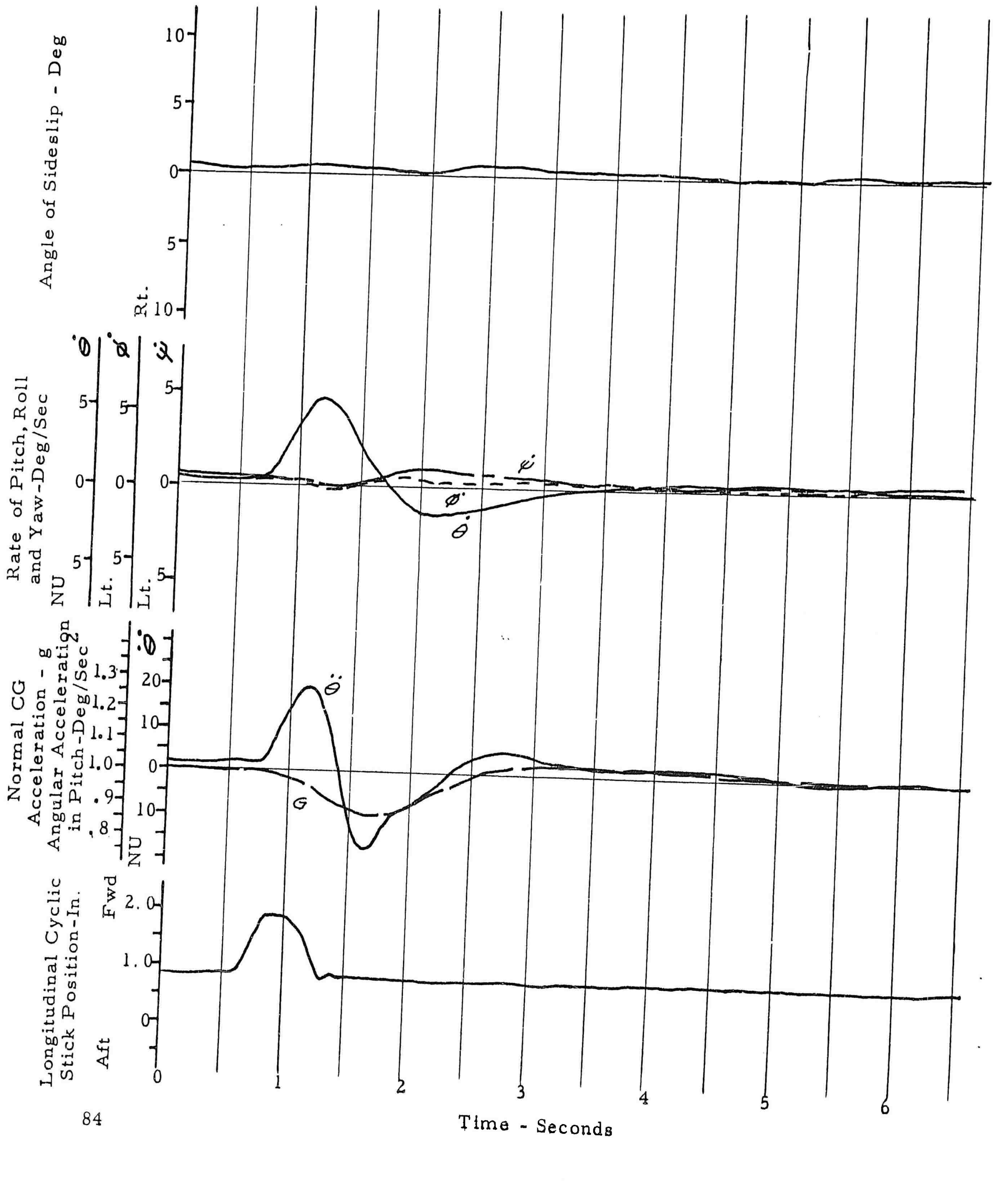


Figure No. 45
DIRECTIONAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 75 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1700 Ft. Rotor RPM = 255
SAS Operating at Full-Gain

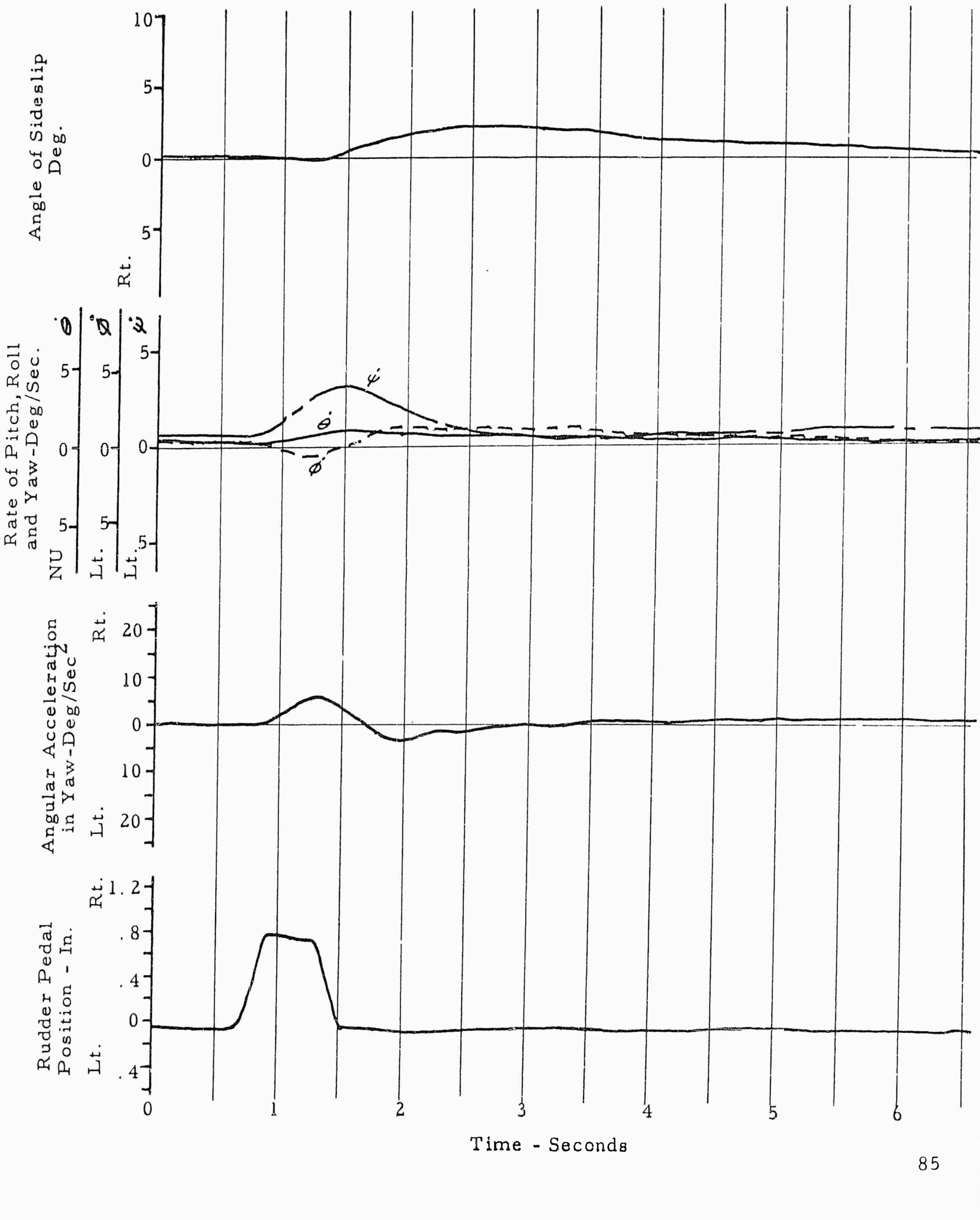


Figure No. 46
DIRECTIONAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 75 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1800 Ft. Rotor RPM = 255
SAS Operating at Full-Gain

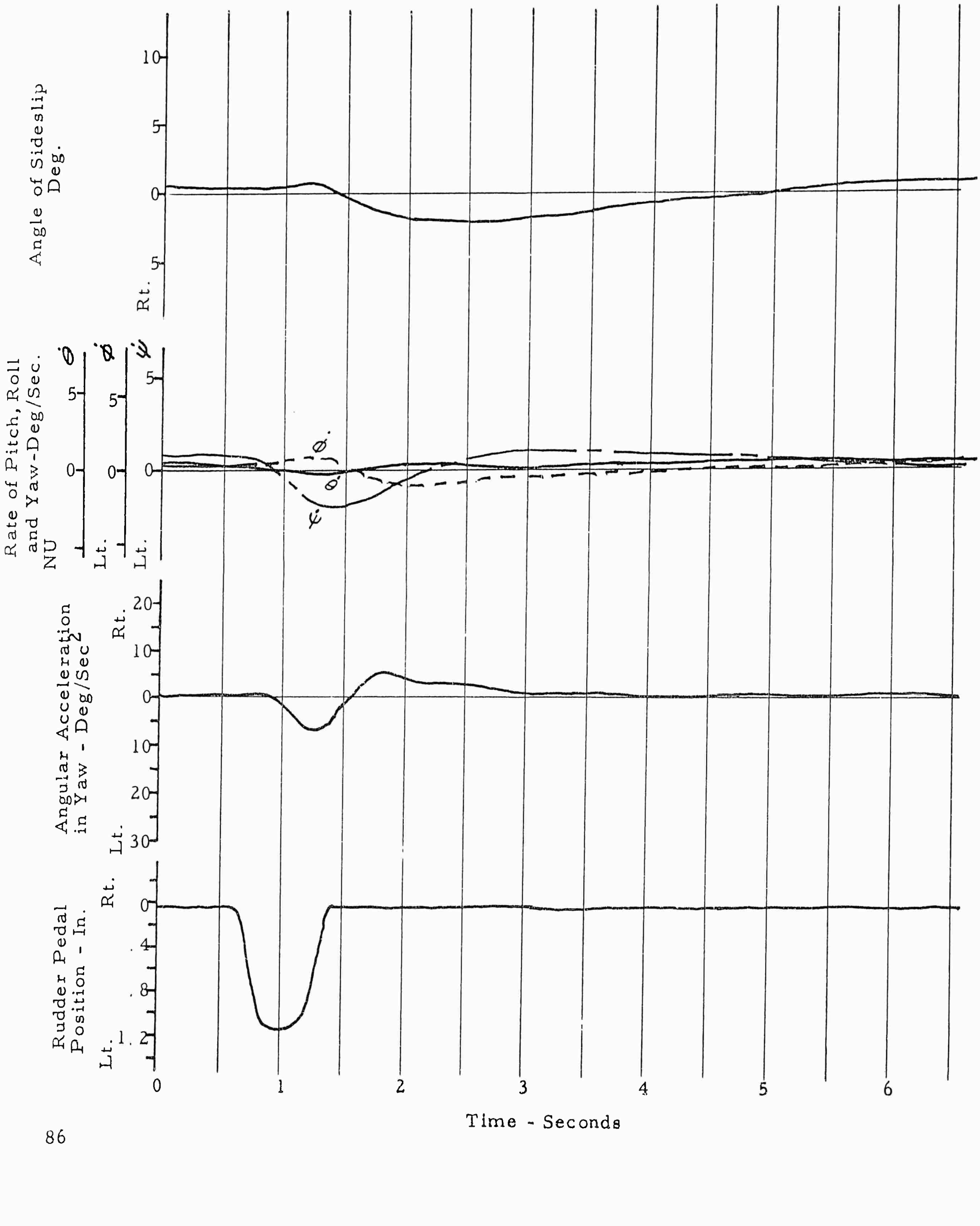


Figure No. 47
LONGITUDINAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 75 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1500 Ft. Rotor RPM = 255
SAS Operating at Half-Gain

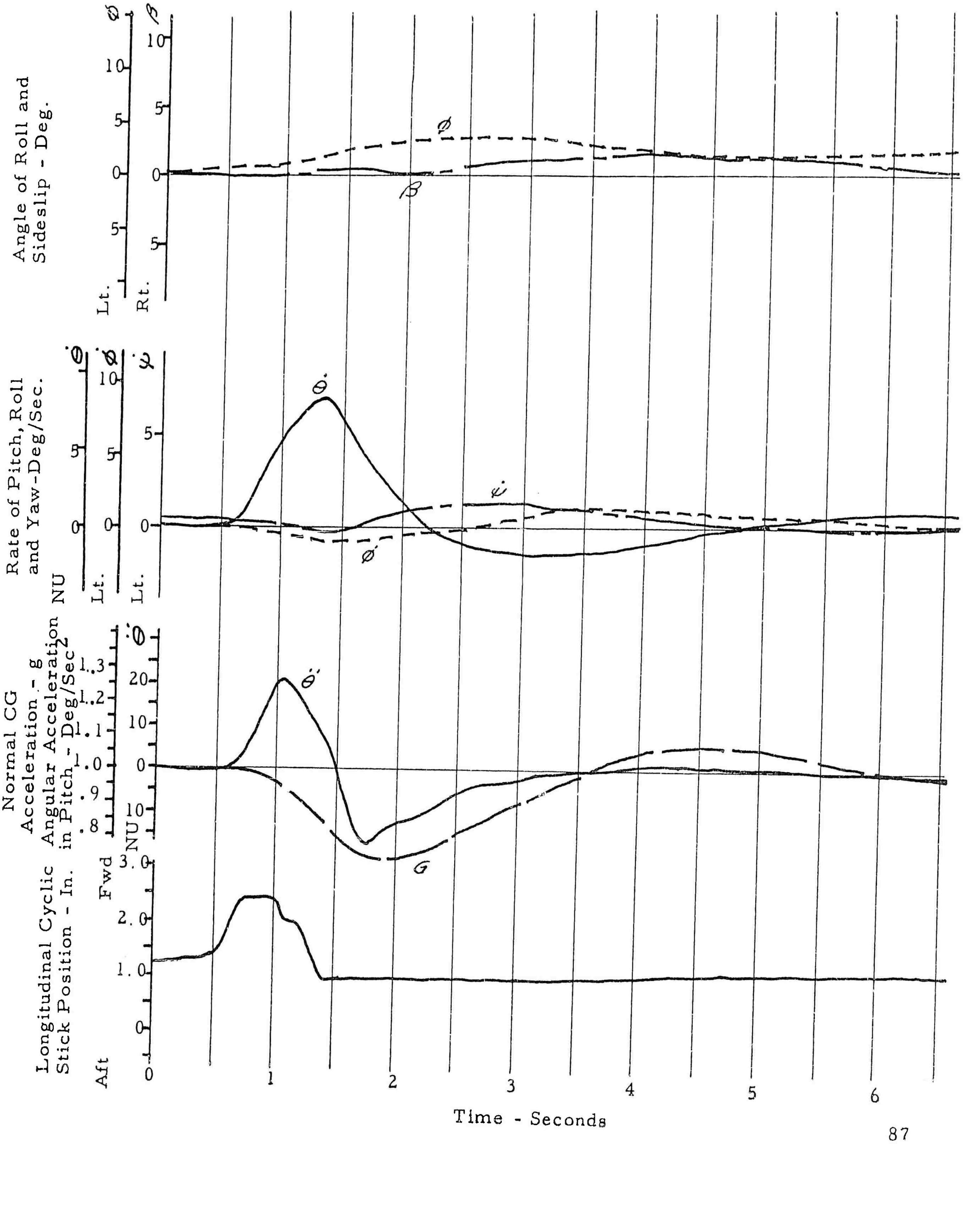


Figure No. 48
LONGITUDINAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 70 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1400 Ft. Rotor RPM = 255
SAS Operating at Half-Gain

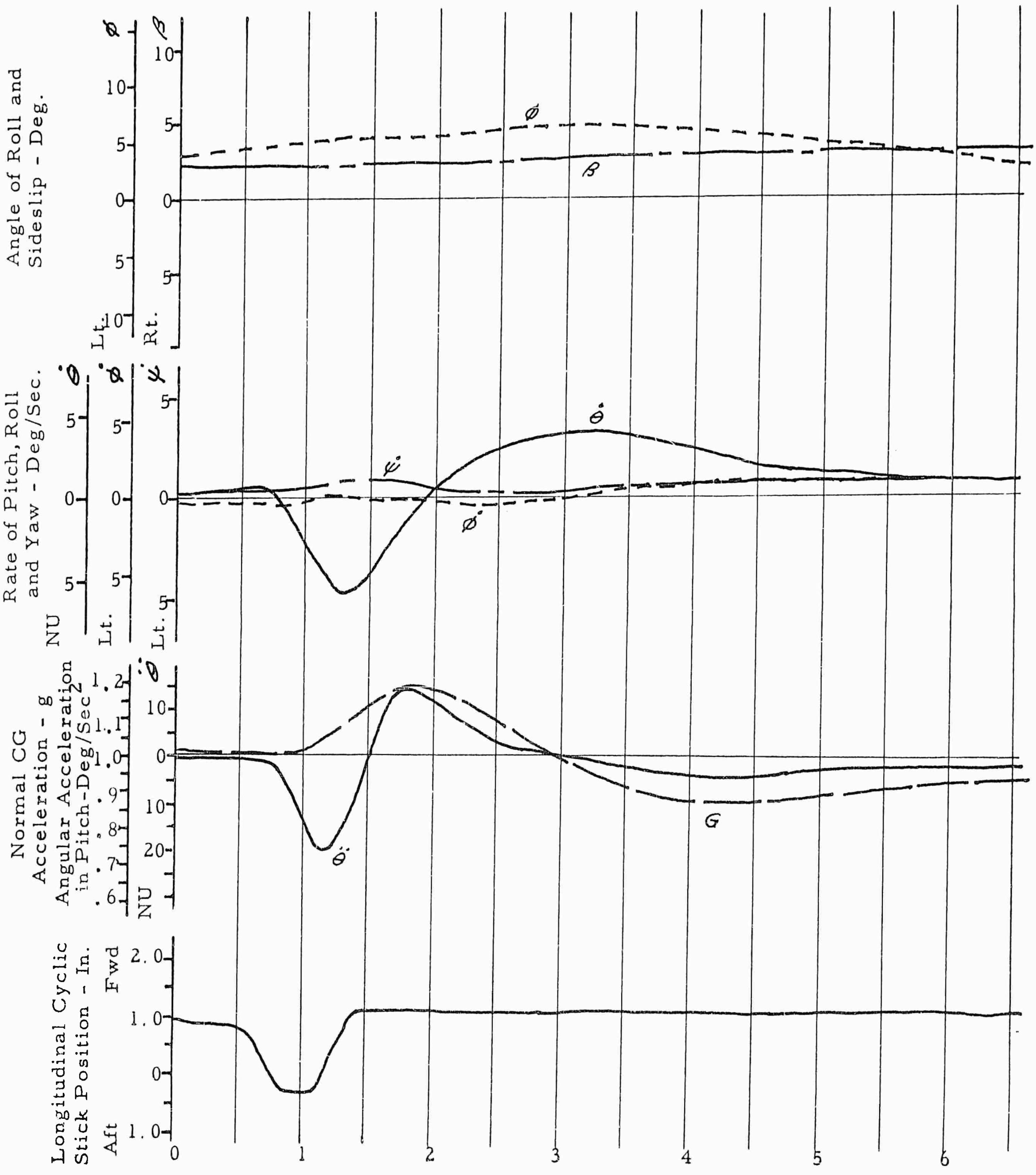


Figure No. 49
DIRECTIONAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 70 Kts. Gross Wt. = 14,800 Lbs.
H_D = 1300 Ft. Rotor RPM = 255
SAS Operating at Half-Gain

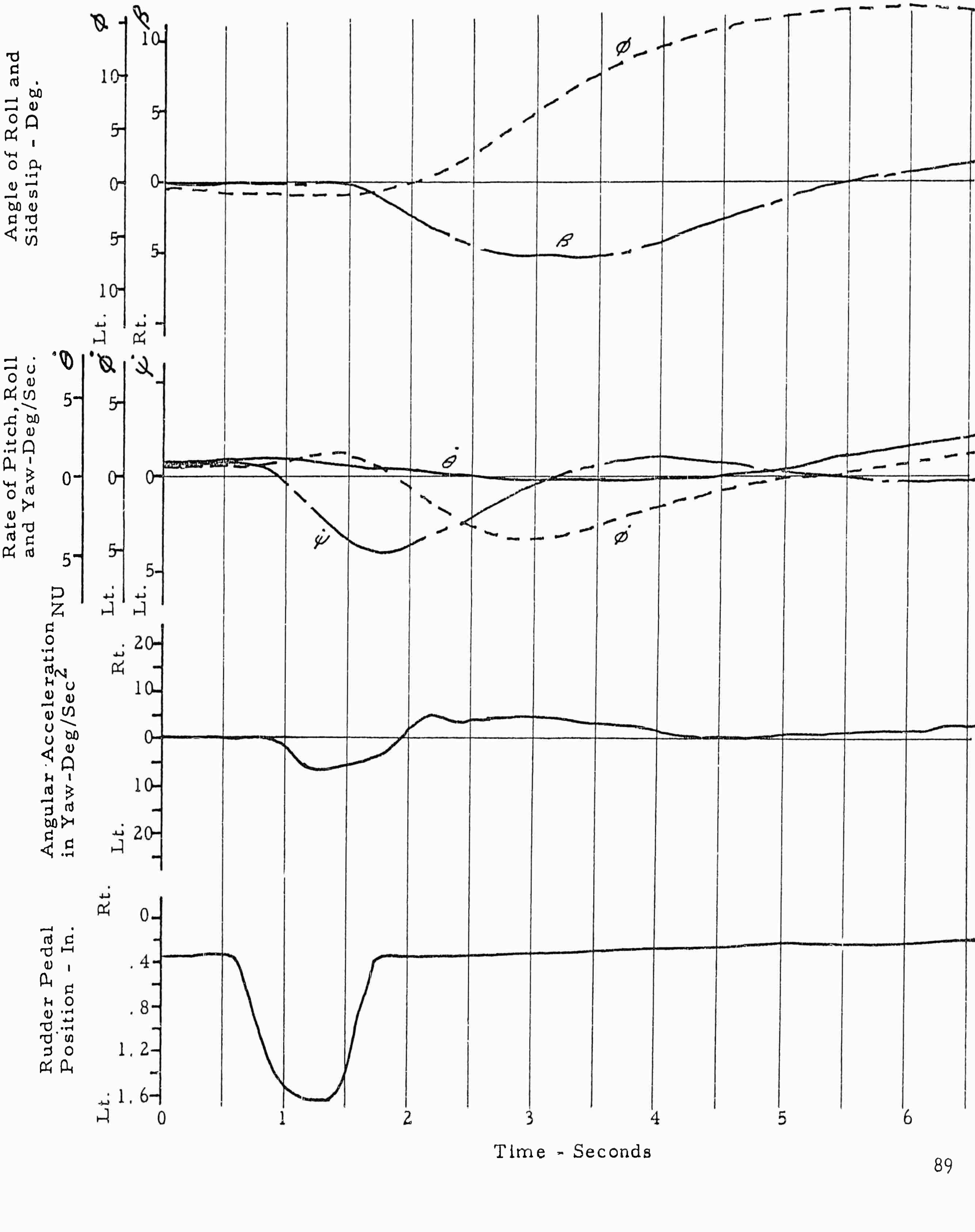


Figure No. 50
DIRECTIONAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 70 Kts. Gross Wt. = 14, 800 Lbs.
H_D = 1000 Ft. Rotor RPM = 255
SAS Operating at Half-Gain

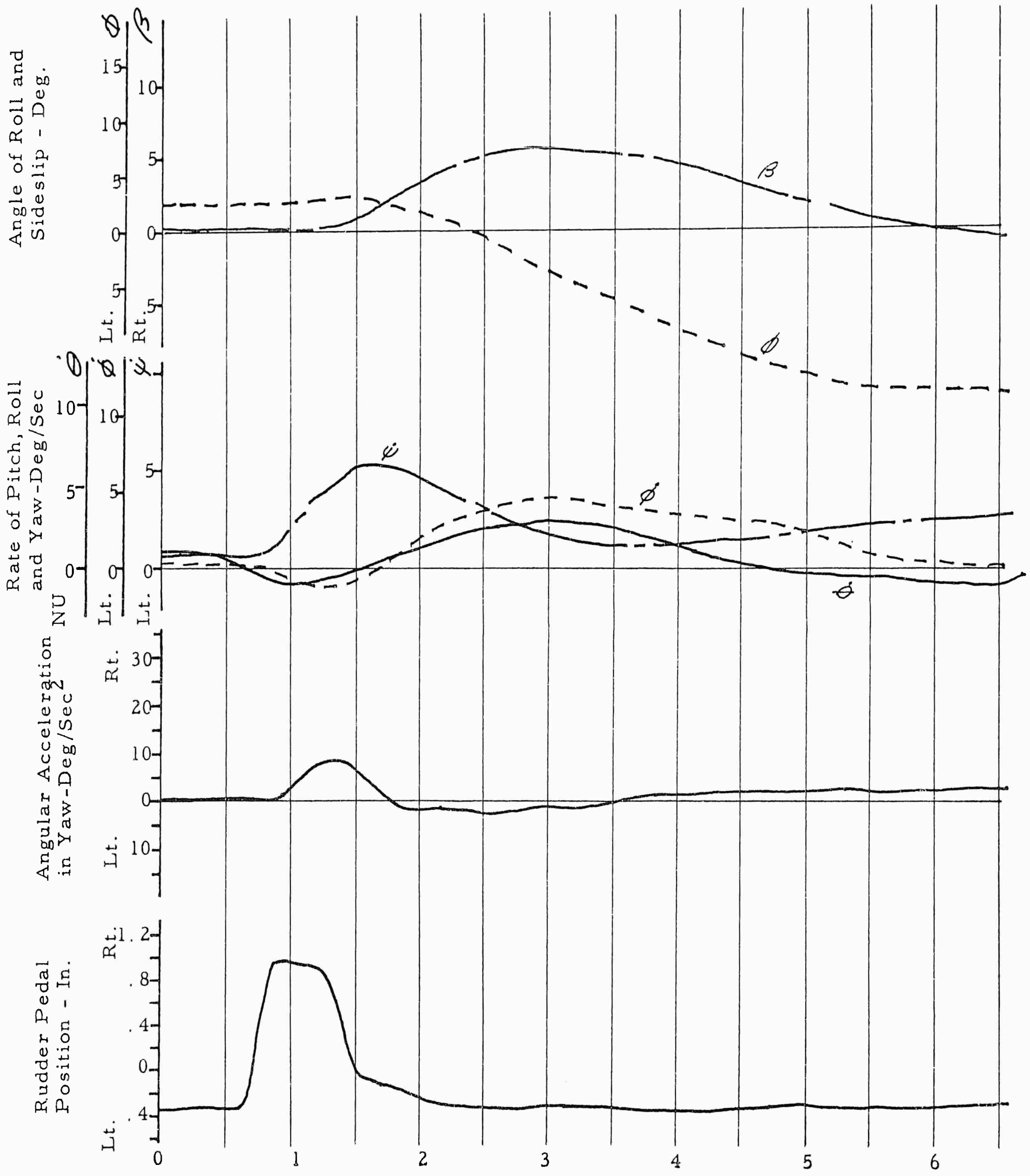


Figure No. 51
LONGITUDINAL DYNAMIC STABILITY
YHC-1A - S/N 58-5514

CAS = 90 Kts. G.W. = 14,400 Lbs.
H_D = 1190 Ft. Rotor RPM = 266

SAS Inoperative

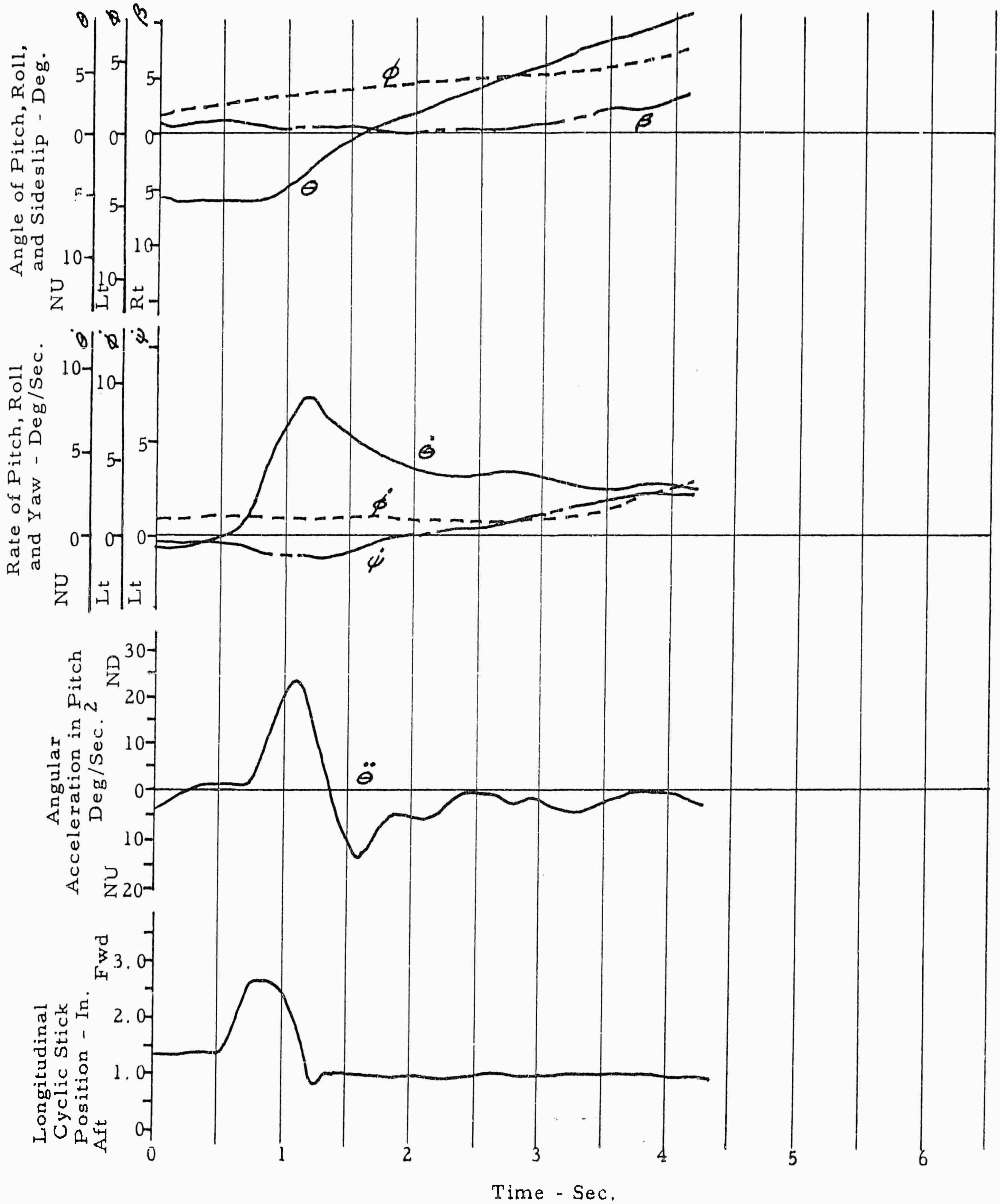


Figure No. 52
LONGITUDINAL DYNAMIC STABILITY
YHC-1A - S/N 58-5514

CAS = 90 Kts. G.W. = 14,400 Lbs.
H_D = 980 Ft. Rotor RPM = 266

SAS Inoperative

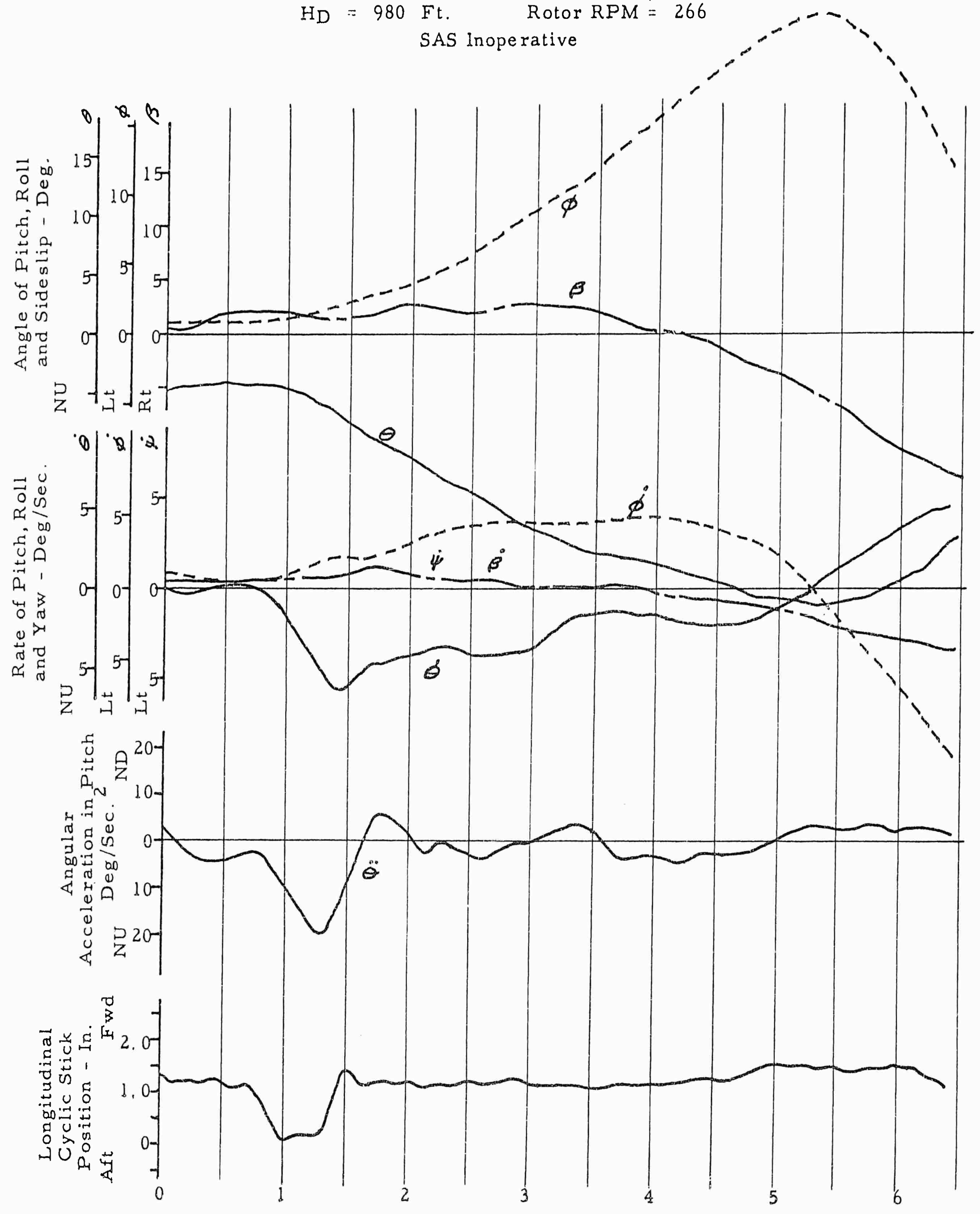


Figure No. 53

LONGITUDINAL DYNAMIC STABILITY

YHC-1A - S/N 58-5514

CAS = 90 Kts. G.W. = 14,400 Lbs.

H_D = 1000 Ft. Rotor RPM = 263

SAS Inoperative

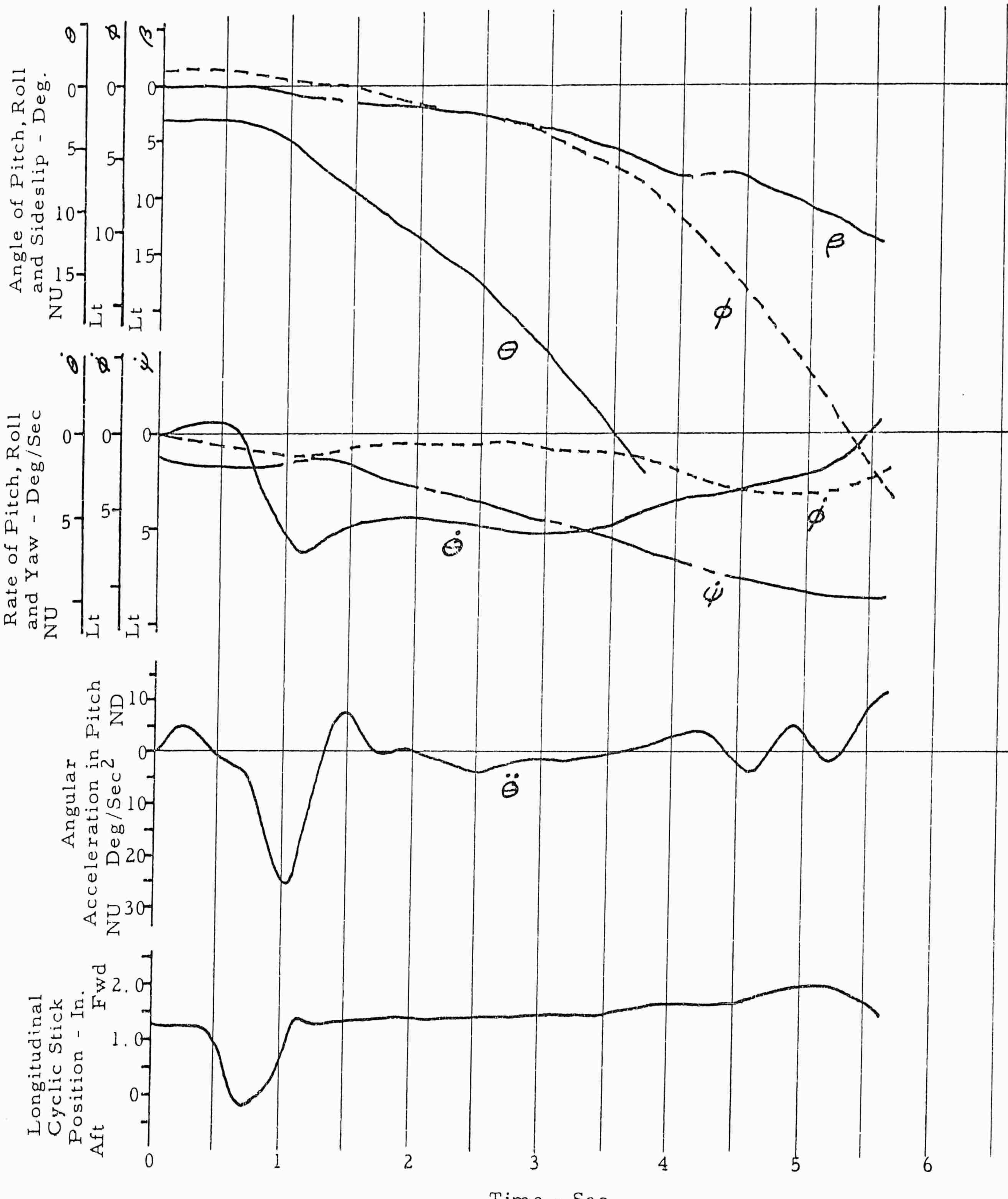


Figure No. 54
DIRECTIONAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 90 Kts.   Gross Wt. = 14,400 Lbs.
HD = 1400 Ft.   Rotor RPM = 267

SAS Inoperative

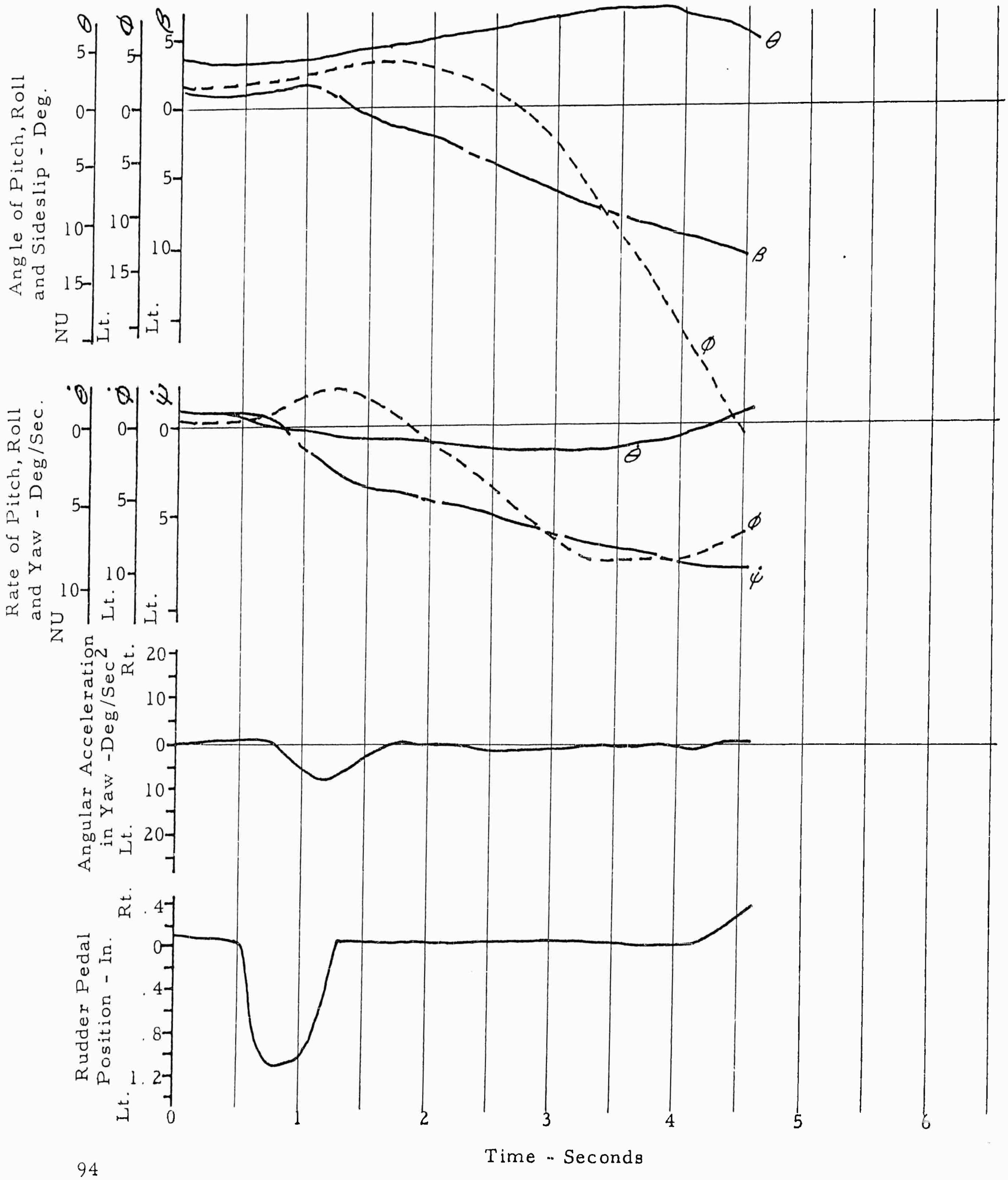


Figure No. 55
DIRECTIONAL DYNAMIC STABILITY
YHC-1A S/N 58-5514

CAS = 90 Kts. Gross Wt. = 14,400 Lbs.
H_D = 1350 Ft. Rotor RPM = 267

SAS Inoperative

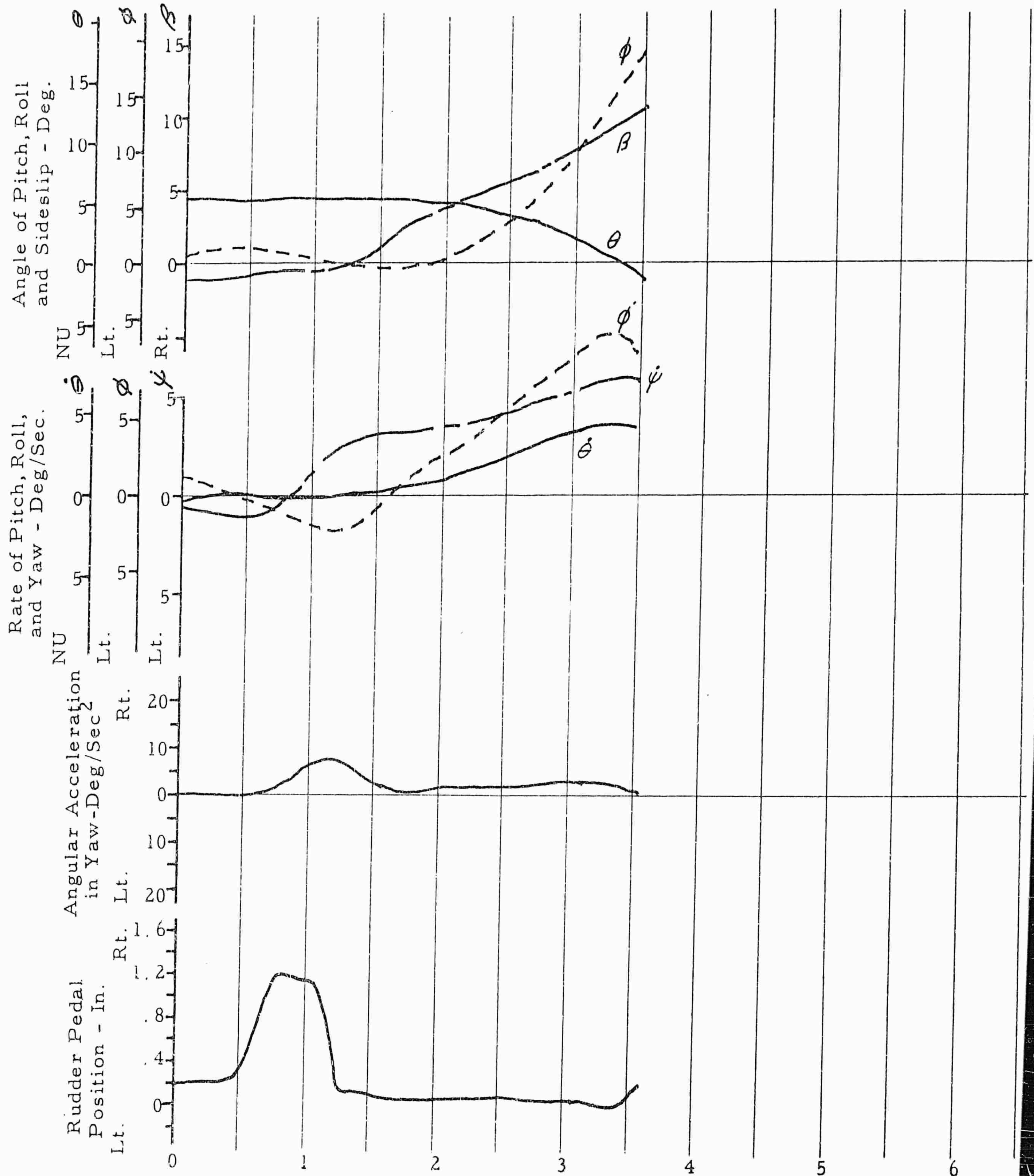


Figure No. 56
LATERAL DYNAMIC STABILITY
YHC-1A - S/N 58-5514

CAS = 90 Kts. G.W. = 14,400 Lbs.
H_D = 1400 Ft. Rotor RPM = 267

SAS Inoperative

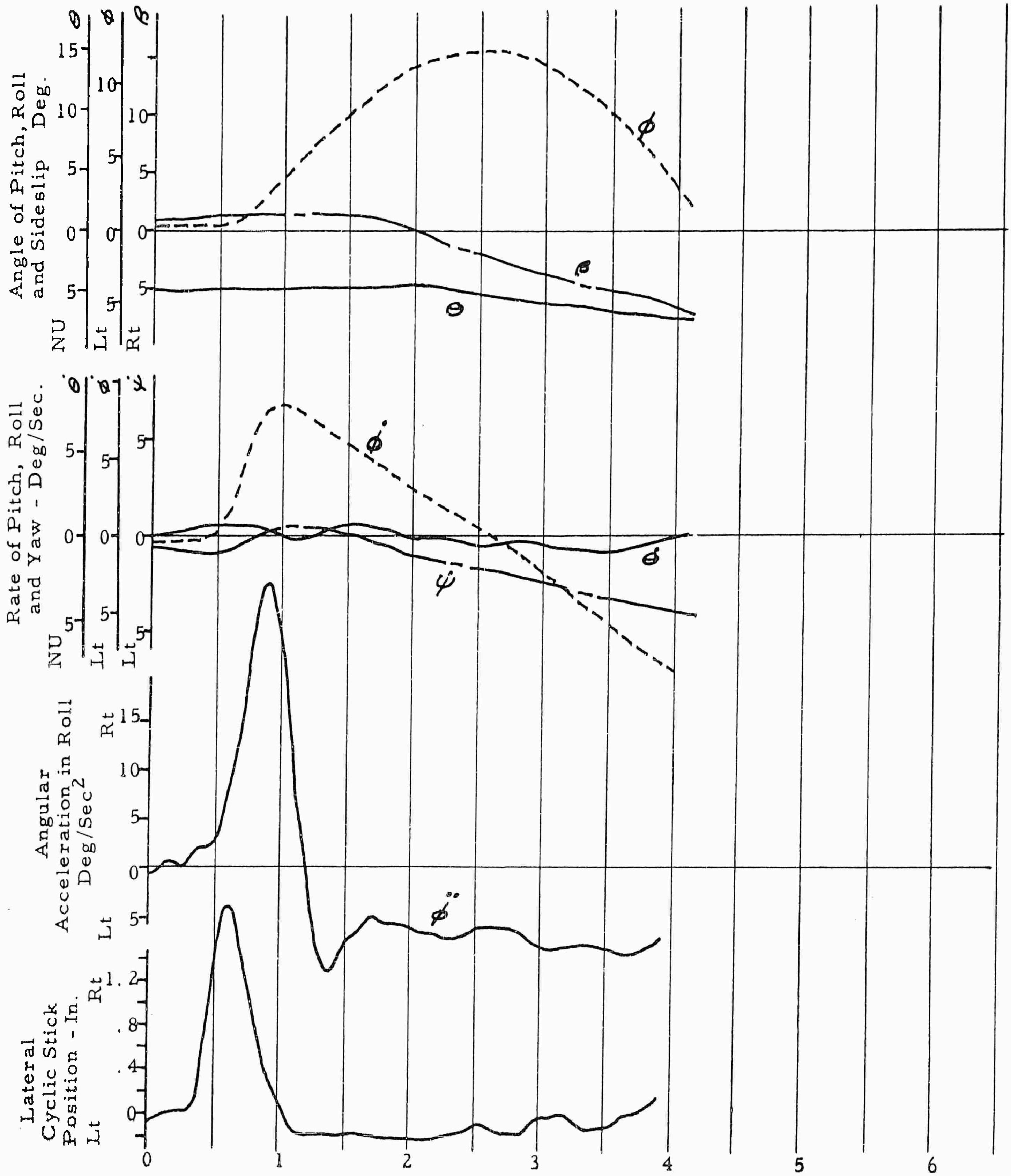
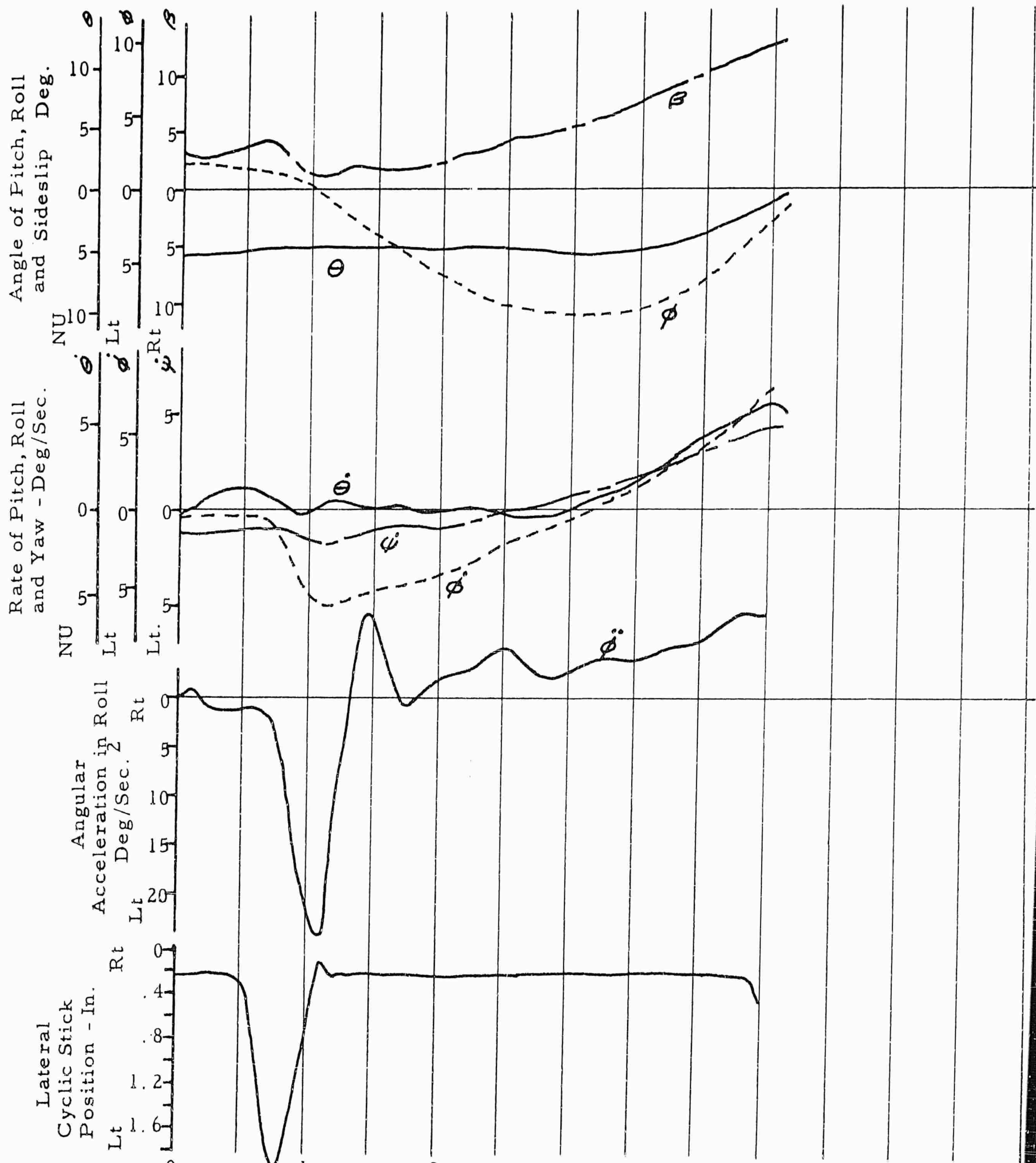


Figure No. 57
LATERAL DYNAMIC STABILITY
YHC -1A - S/N 58-5514

CAS = 90 Kts. G.W. = 14,400 Lbs.
H_D = 1430 Ft. Rotor RPM = 267

SAS Inoperative



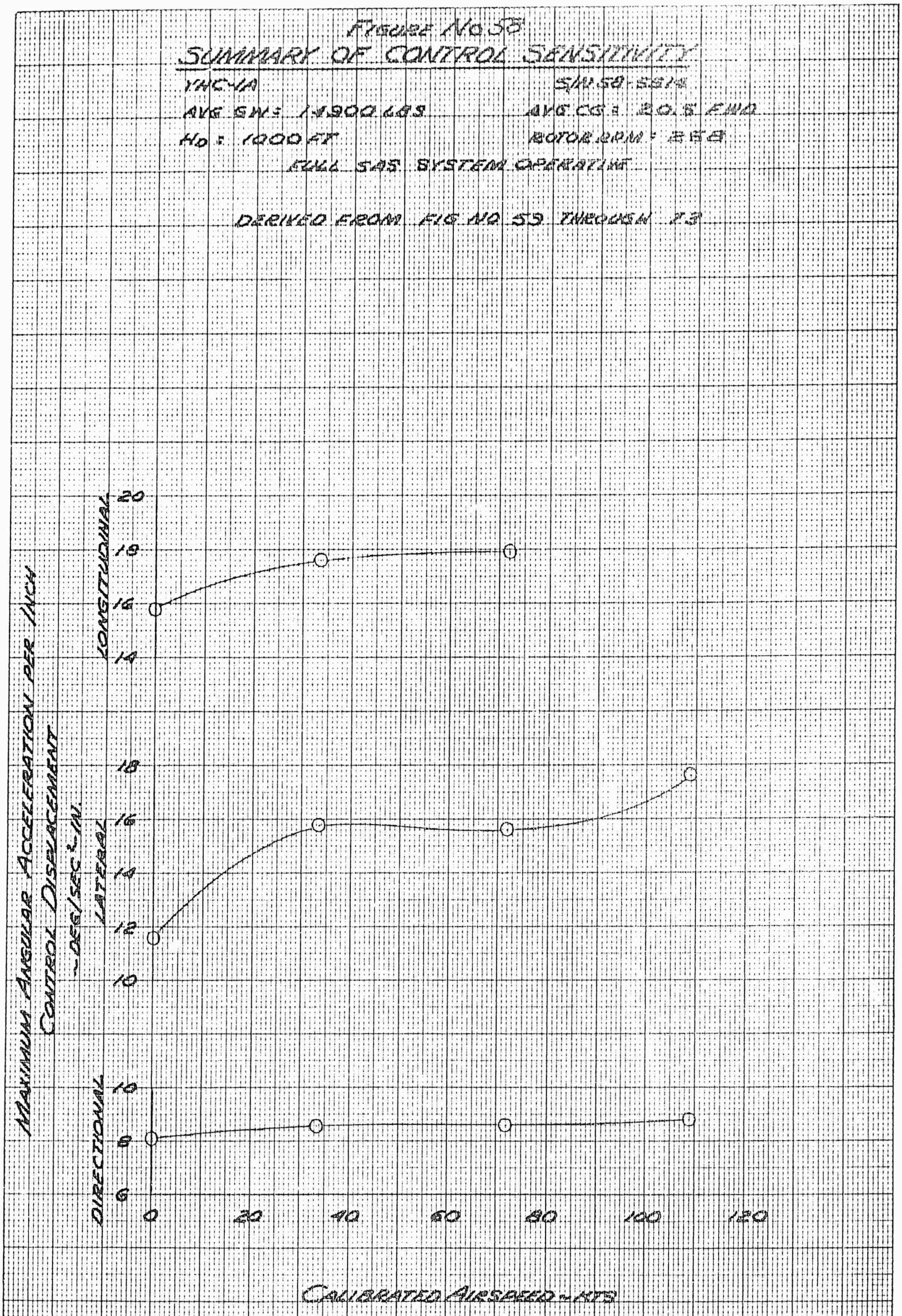

SUMMARY OF CONTROL SENSITIVITY
YHC-1A
AVG GW = 14900 LBS
AVG CG = 20.5 FWD
H0 = 1000 FT
FULL SAS SYSTEM OPERATING
MAXIMUM ANGULAR ACCELERATION PER INCH
CONTROL DISPLACEMENT
~DEG/SEC²/IN.
LONGITUDINAL
LATERAL
DIRECTIONAL
20
18
16
14
18
16
14
12
10
10
8
6
0
20
40
60
80
100
120
CALIBRATED AIRSPEED ~ KTS

FIGURE No 59

LONGITUDINAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

FULL SAS OPERATING

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 14500 LBS AVG CG = 25.0 IN. FWD

ROTOR RPM = 258

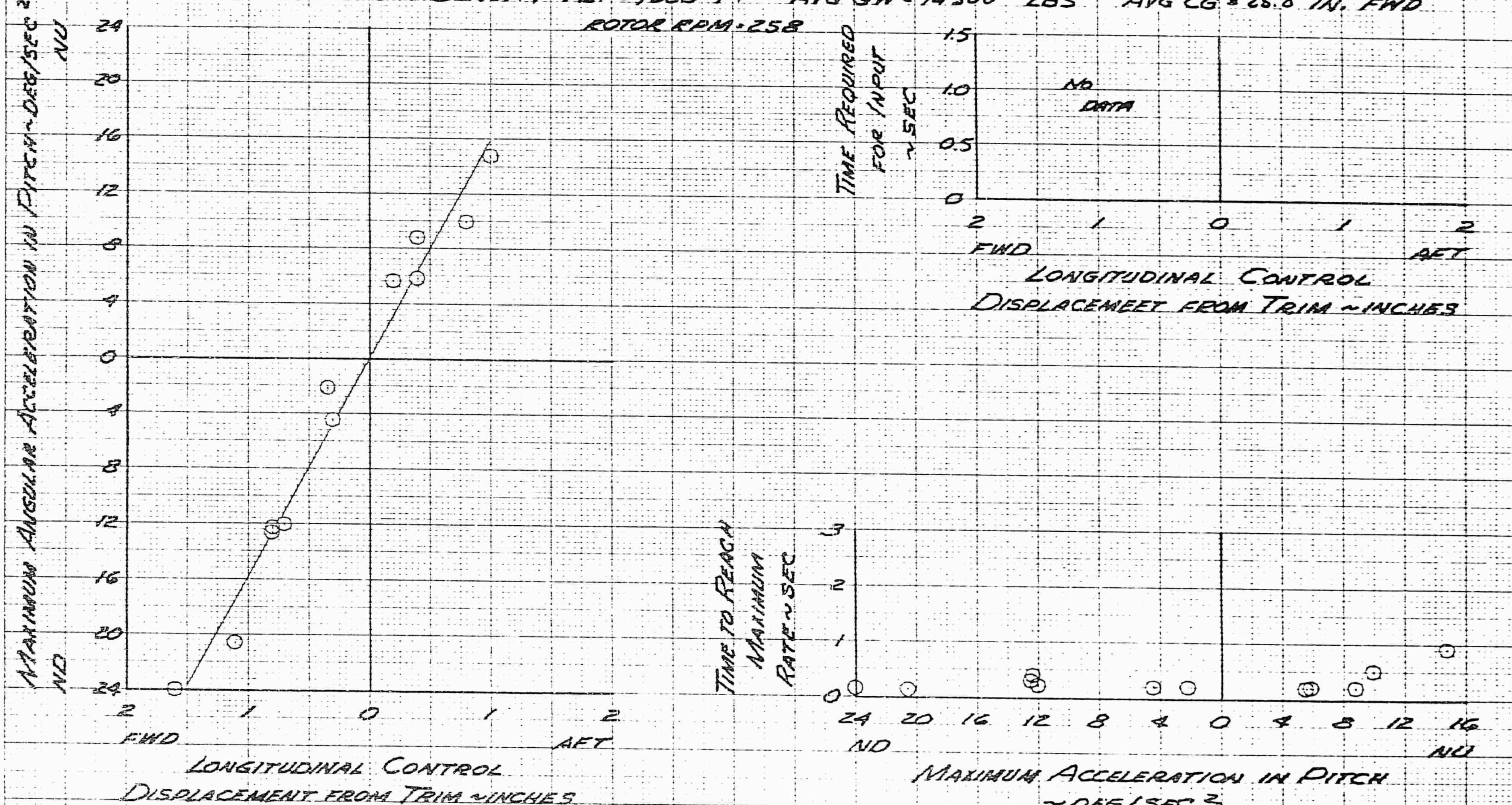


FIGURE No 60

LONGITUDINAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 33.5 KTS DENSITY ALT = 1000 FT AVG GW = 15250 LBS AVG CG = 221 IN. FWD

ROTOR RPM = 258

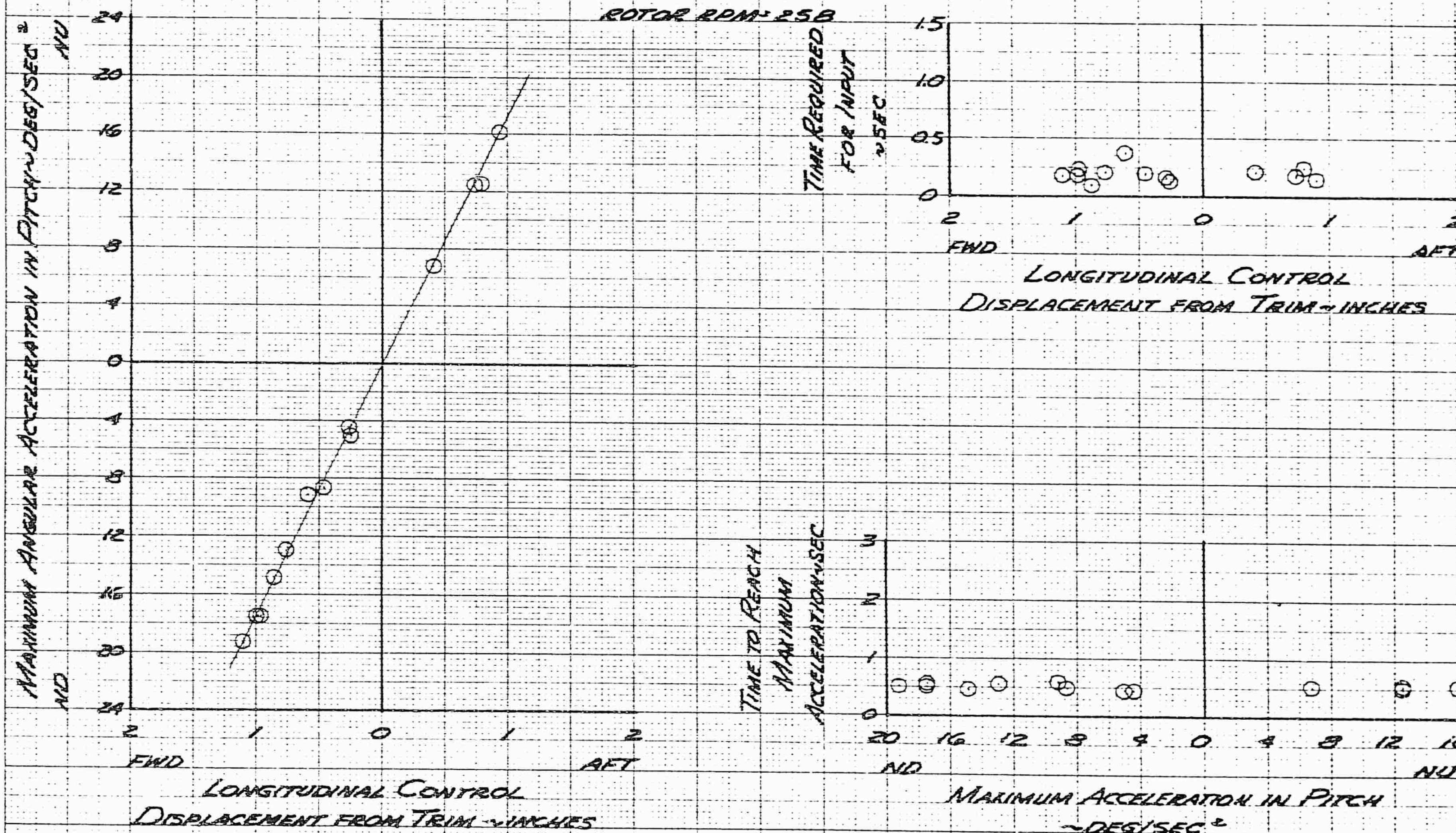


FIGURE NO 61

LONGITUDINAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 72 KTS DENSITY ALT = [illegible] FT AVG GW = 15100 LBS AVG CG = 226 IN. FWD

ROTOR RPM = 258

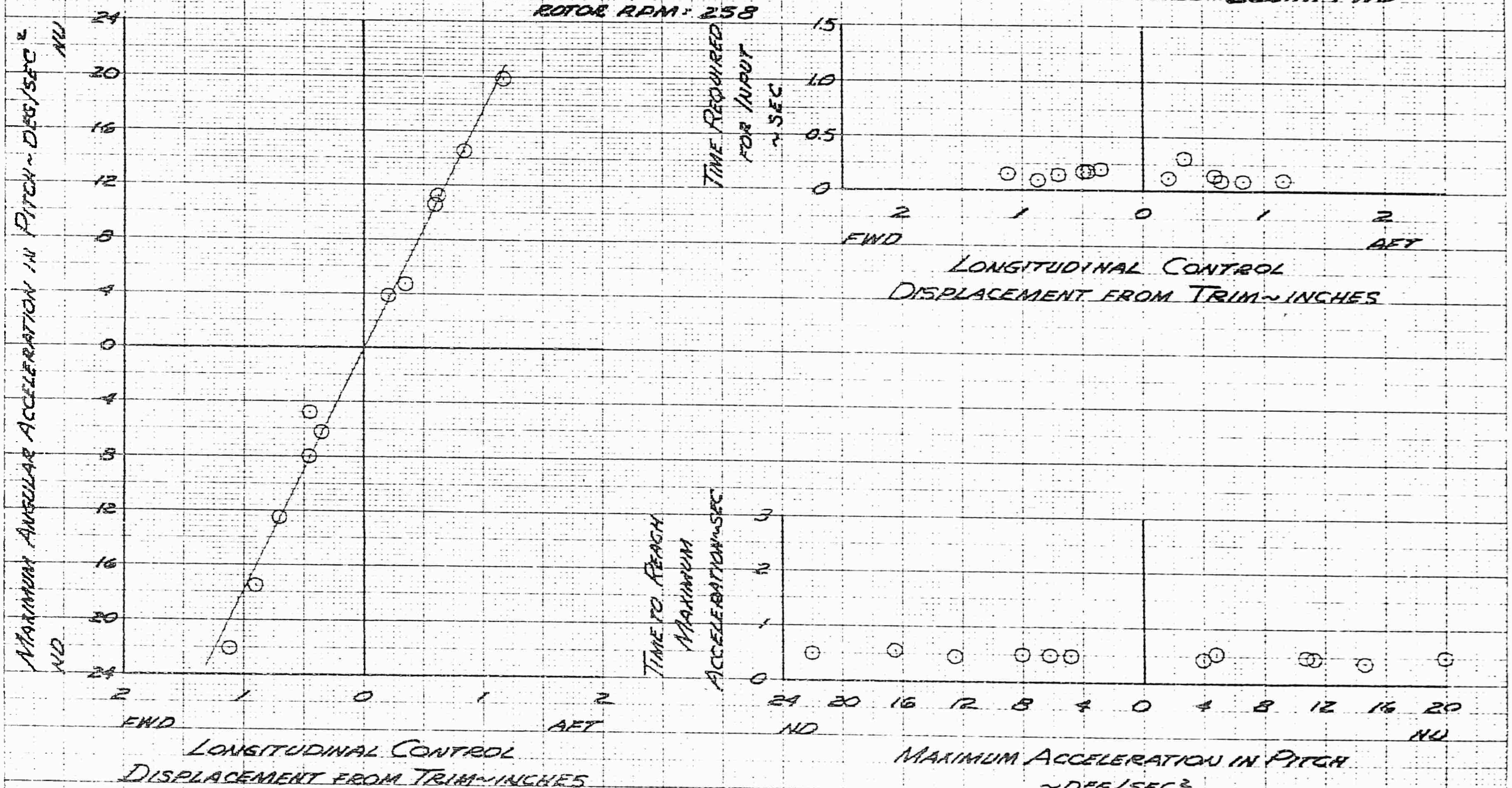


FIGURE NO 62

LATERAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 15000 LBS AVG CG = 25.0 IN. FWD

ROTOR RPM = 258

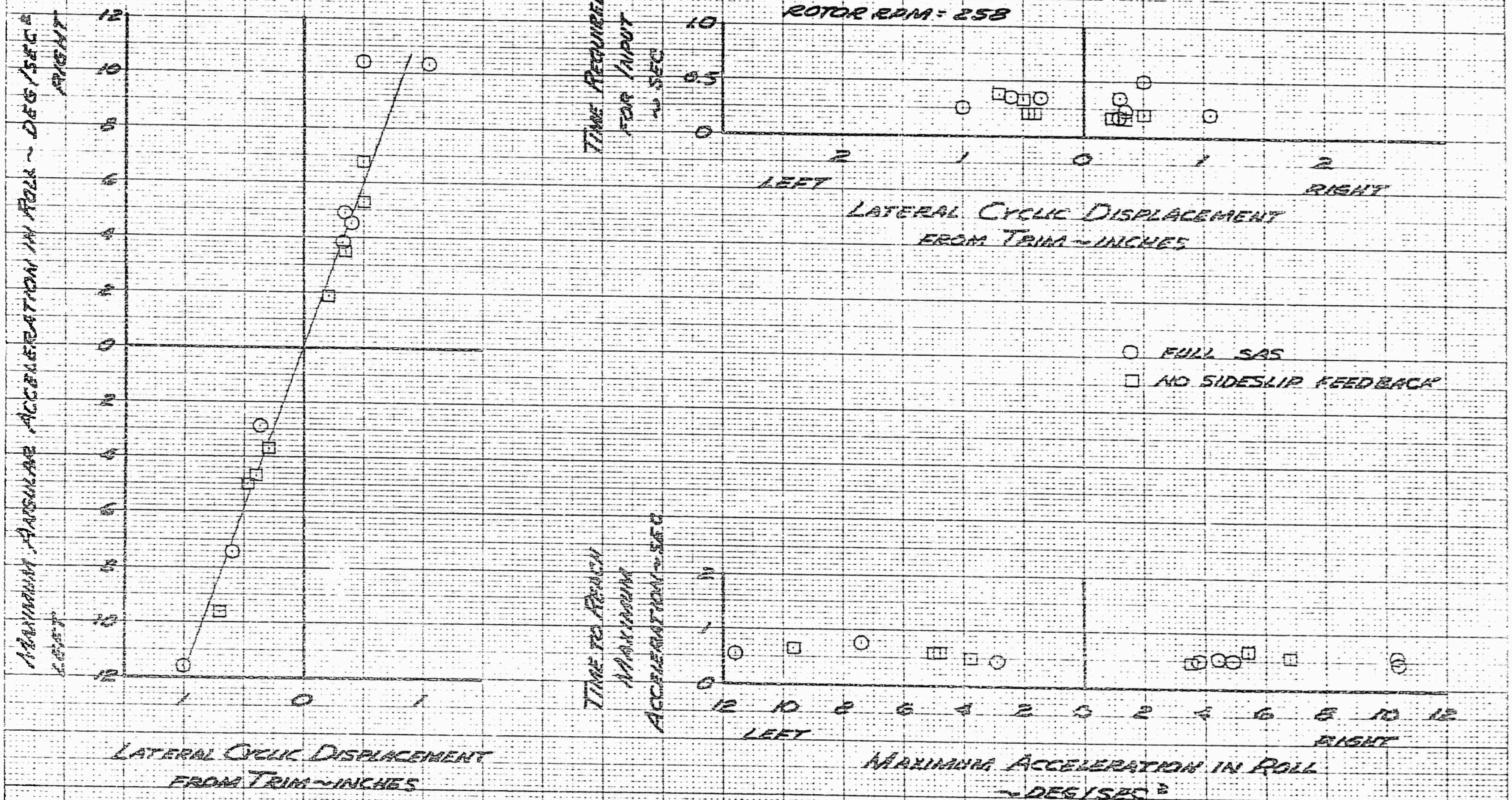


FIGURE NO 63

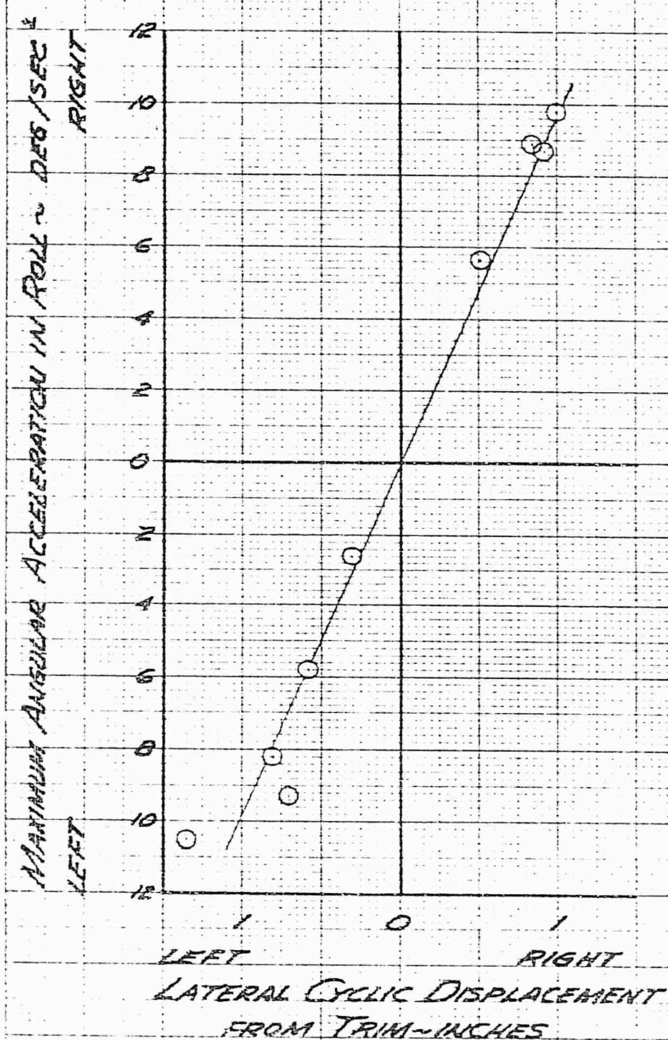
LATERAL CONTROL SENSITIVITY

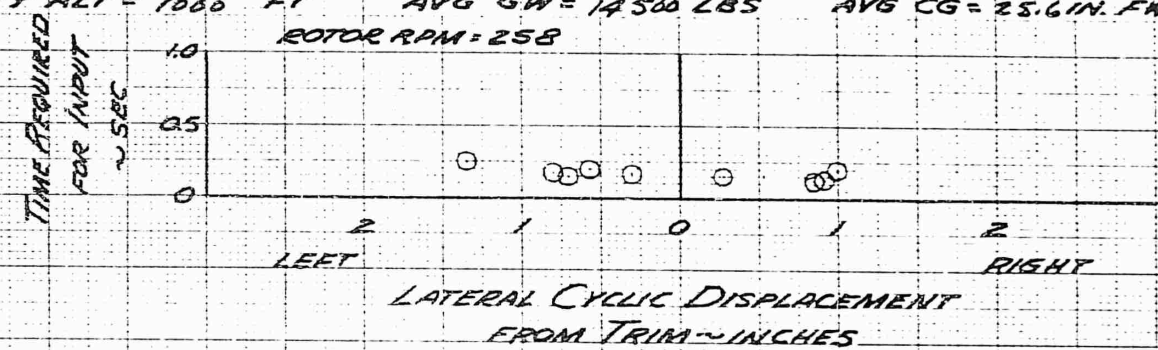
YHC-1A S/N 58-5514

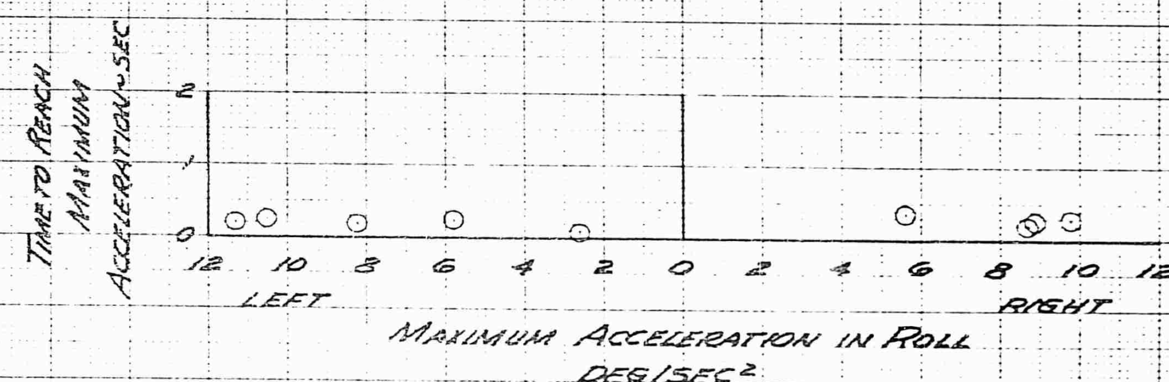
SAS SYSTEM OPERATING AT HALF GAIN

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 14500 LBS AVG CG = 25.6 IN. FWD

ROTOR RPM = 258







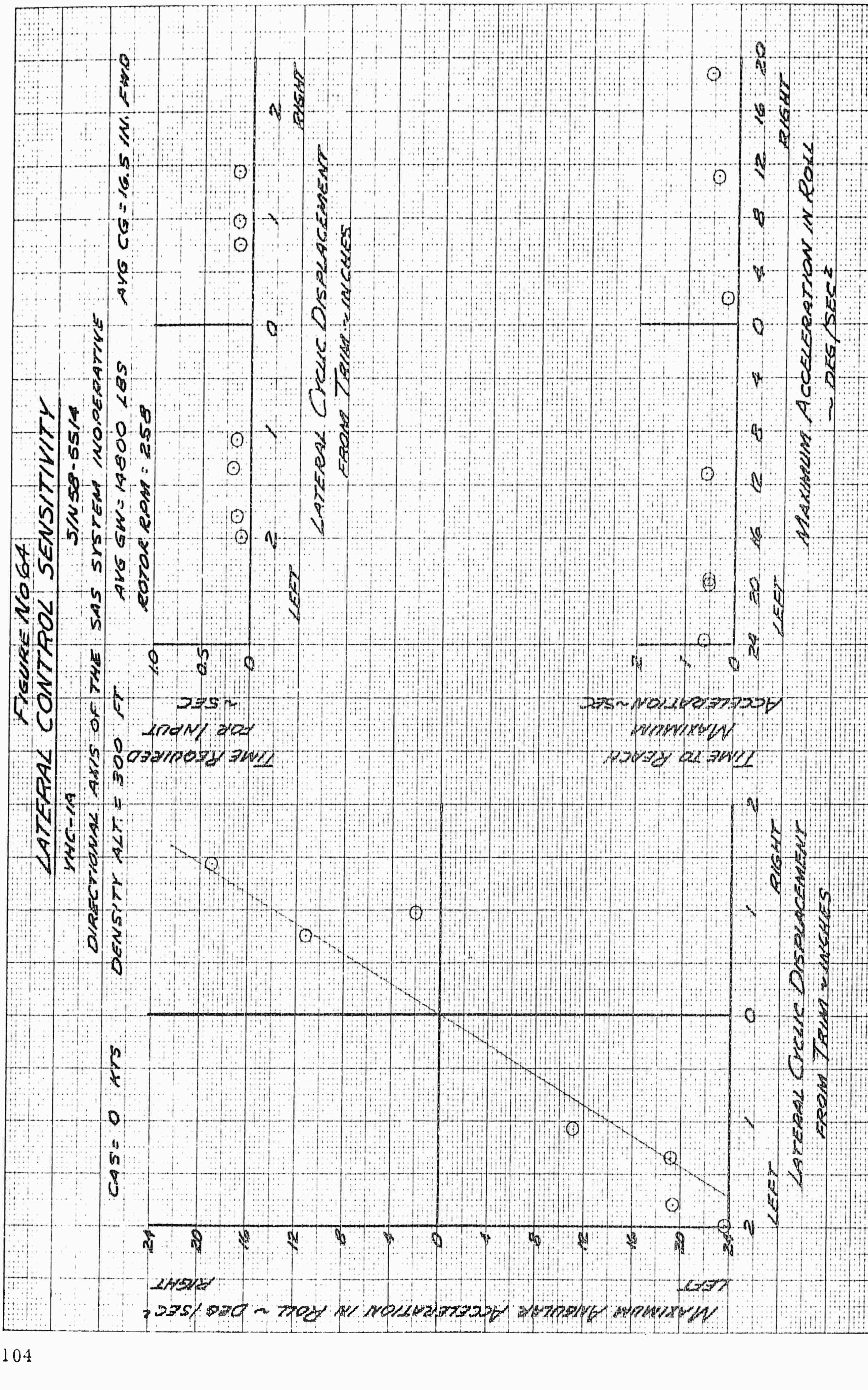


FIGURE NO 64
LATERAL CONTROL SENSITIVITY
YHC-1A
S/N 58-5514
DIRECTIONAL AXIS OF THE SAS SYSTEM INOPERATIVE
CAS = 0 KTS
DENSITY ALT. = 300 FT
AVG GW = 14800 LBS
AVG CG = 16.5 IN. FWD
ROTOR RPM = 258
TIME REQUIRED FOR INPUT ~ SEC
LATERAL CYCLIC DISPLACEMENT FROM TRIM ~ INCHES
LEFT
RIGHT
TIME TO REACH MAXIMUM ACCELERATION ~ SEC
MAXIMUM ACCELERATION IN ROLL ~ DEG/SEC²
MAXIMUM ANGULAR ACCELERATION IN ROLL ~ DEG/SEC²

FIGURE No 65

LATERAL CONTROL SENSITIVITY

YHC-1A SIN 58-5514

CAS = 33.5 KTS

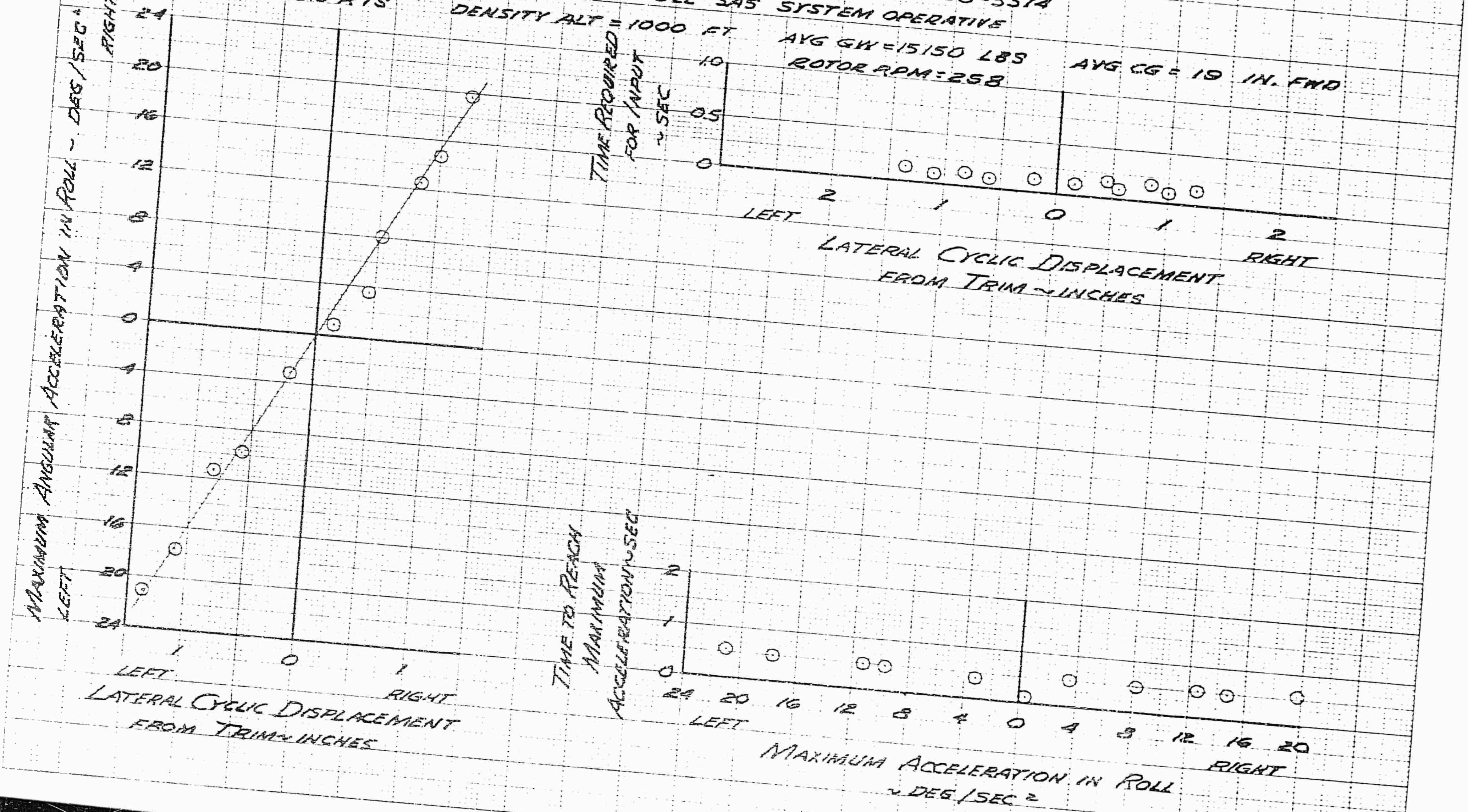
FULL SAS SYSTEM OPERATIVE

DENSITY ALT = 1000 FT

AVG GW = 15150 LBS

ROTOR RPM = 258

AVG CG = 19 IN. FWD



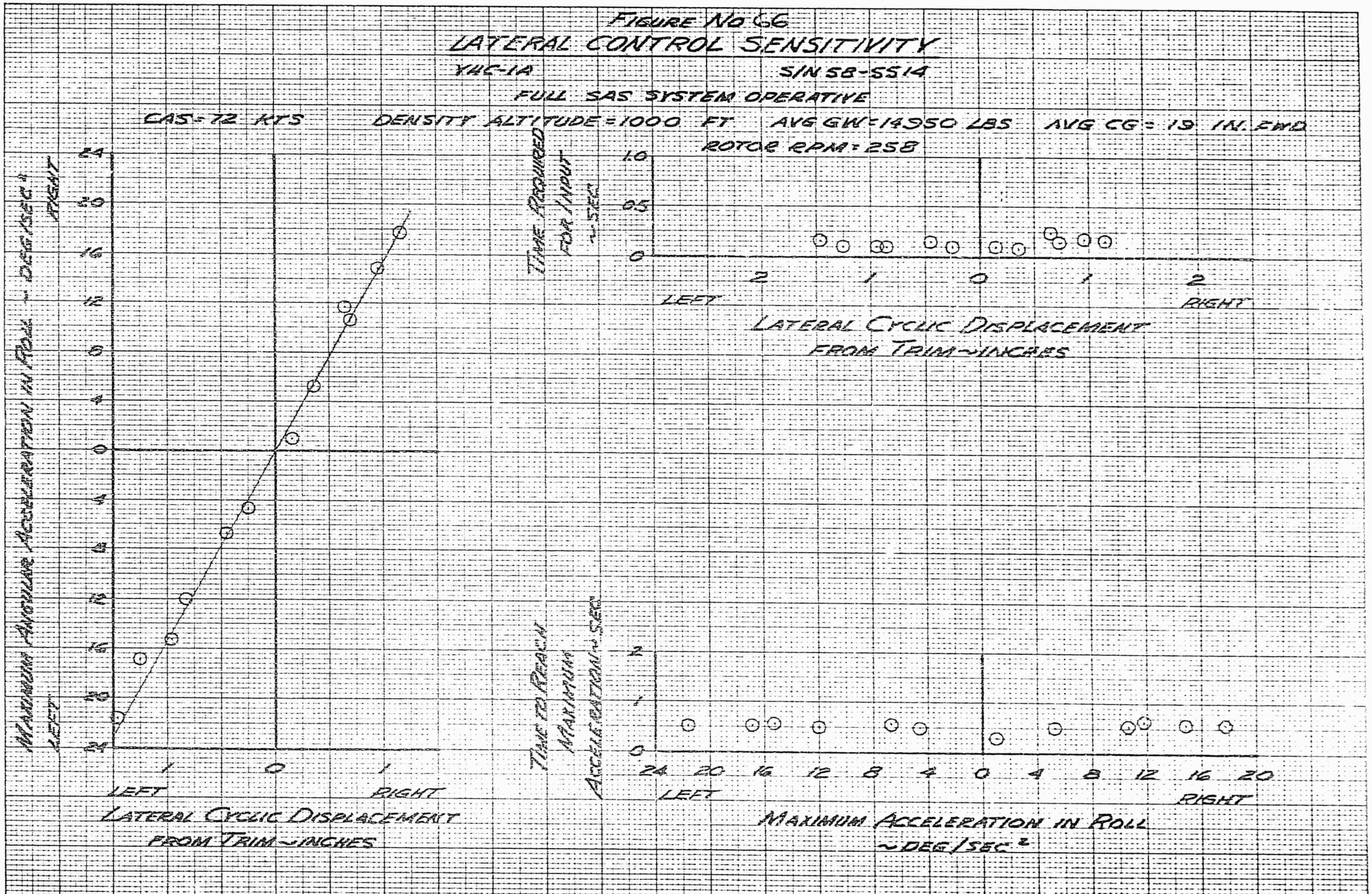


FIGURE NO 66
LATERAL CONTROL SENSITIVITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
CAS = 72 KTS
DENSITY ALTITUDE = 1000 FT
AVG GW = 14950 LBS
AVG CG = 19 IN. FWD
ROTOR RPM = 258
MAXIMUM ANGULAR ACCELERATION IN ROLL ~ DEG/SEC²
RIGHT
LEFT
24
20
16
12
8
4
0
4
8
12
16
20
24
1
0
1
LEFT
RIGHT
LATERAL CYCLIC DISPLACEMENT FROM TRIM ~ INCHES
TIME REQUIRED FOR INPUT ~ SEC
1.0
.05
0
2
1
0
1
2
LEFT
RIGHT
LATERAL CYCLIC DISPLACEMENT FROM TRIM ~ INCHES
TIME TO REACH MAXIMUM ACCELERATION ~ SEC
2
1
0
24
20
16
12
8
4
0
4
8
12
16
20
LEFT
RIGHT
MAXIMUM ACCELERATION IN ROLL ~ DEG/SEC²

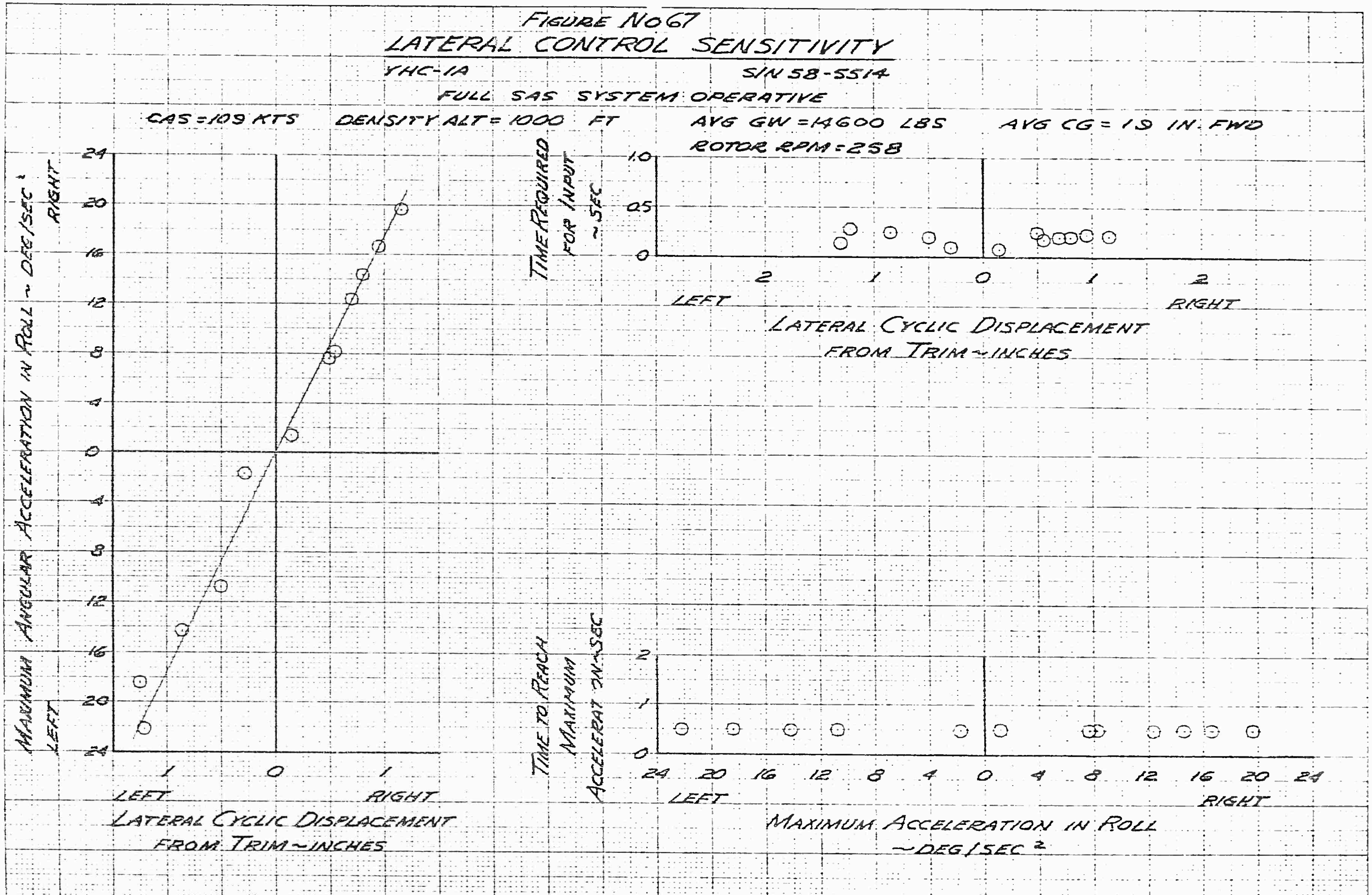


FIGURE No 67
LATERAL CONTROL SENSITIVITY
YHC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
CAS = 109 KTS
DENSITY ALT = 1000 FT
AVG GW = 14600 LBS
AVG CG = 19 IN. FWD
ROTOR RPM = 258
MAXIMUM ANGULAR ACCELERATION IN ROLL ~ DEG/SEC²
RIGHT
LEFT
24
20
16
12
8
4
0
4
8
12
16
20
24
1
0
1
LEFT
RIGHT
LATERAL CYCLIC DISPLACEMENT FROM TRIM ~ INCHES
TIME REQUIRED FOR INPUT ~ SEC
1.0
0.5
0
2
1
0
1
2
LEFT
RIGHT
LATERAL CYCLIC DISPLACEMENT FROM TRIM ~ INCHES
TIME TO REACH MAXIMUM ACCELERATION ~ SEC
2
1
0
24
20
16
12
8
4
0
4
8
12
16
20
24
LEFT
RIGHT
MAXIMUM ACCELERATION IN ROLL ~ DEG/SEC²

FIGURE No 68
DIRECTIONAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 15050 LBS AVG CG = 19 IN. FWD
ROTOR RPM = 258

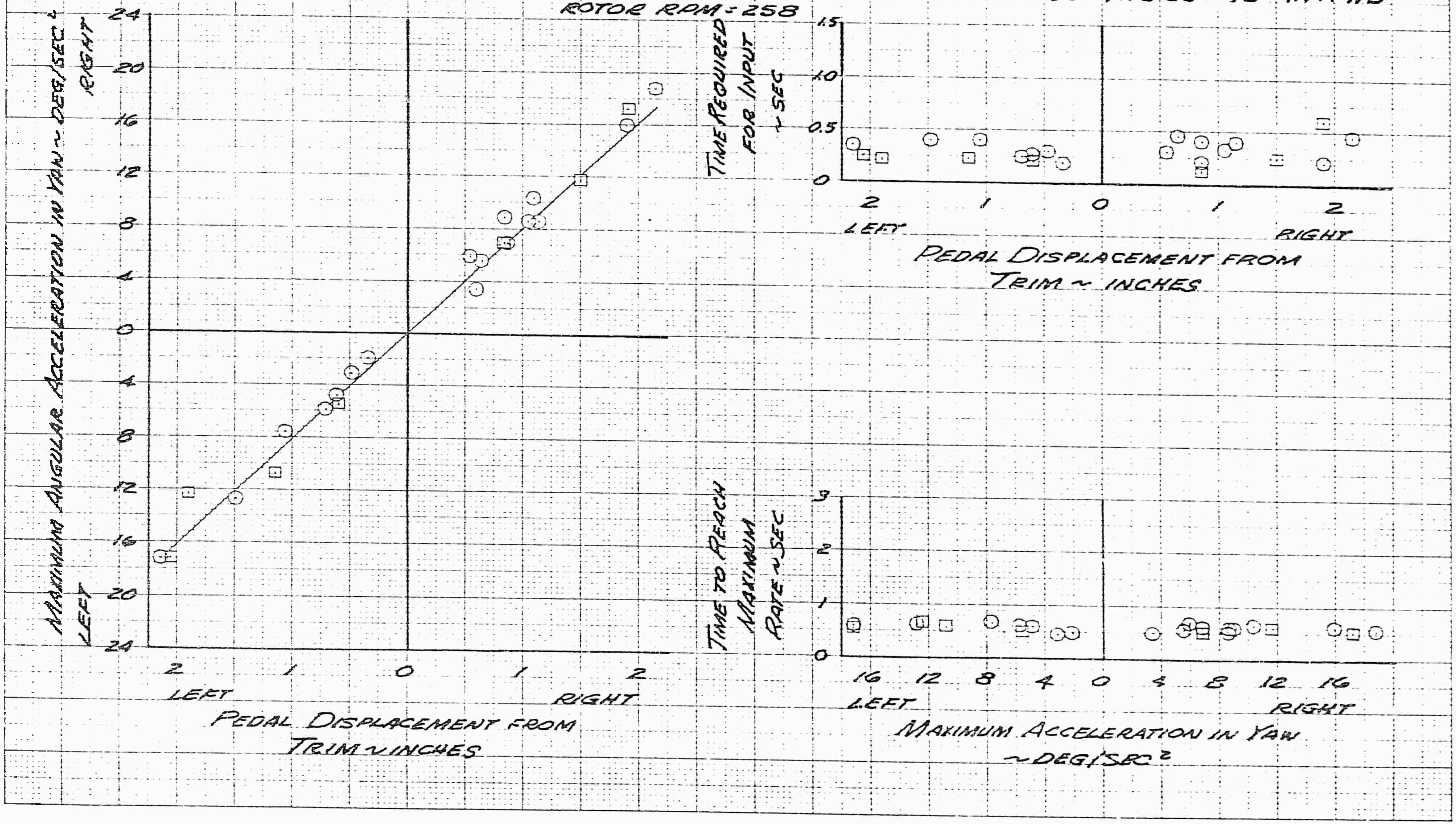


FIGURE No 69

DIRECTIONAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

SAS OPERATING AT HALF GAIN

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 14500 LBS AVG CG = 256 IN. FWD

ROTOR RPM = 258

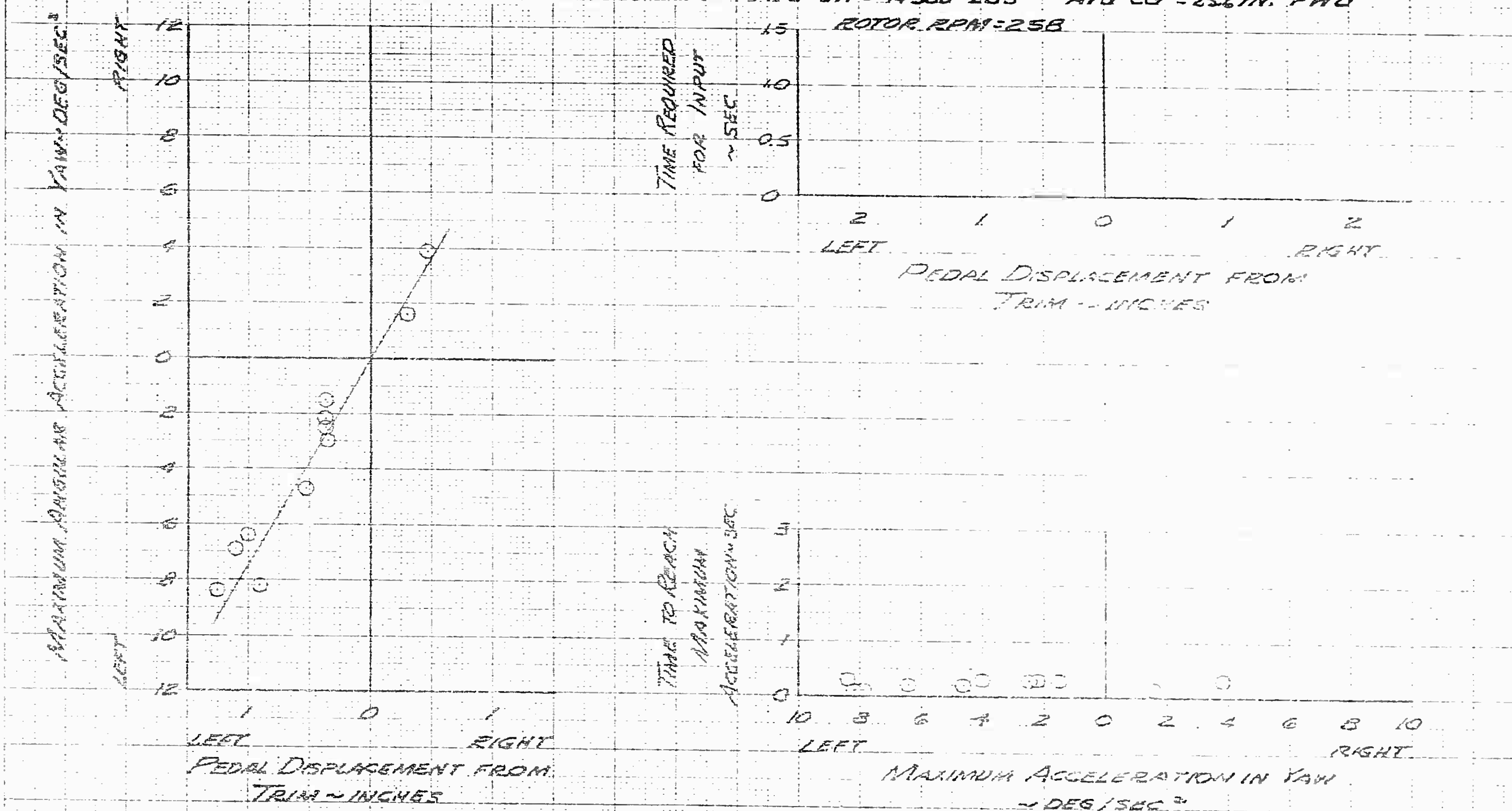


FIGURE NO 70

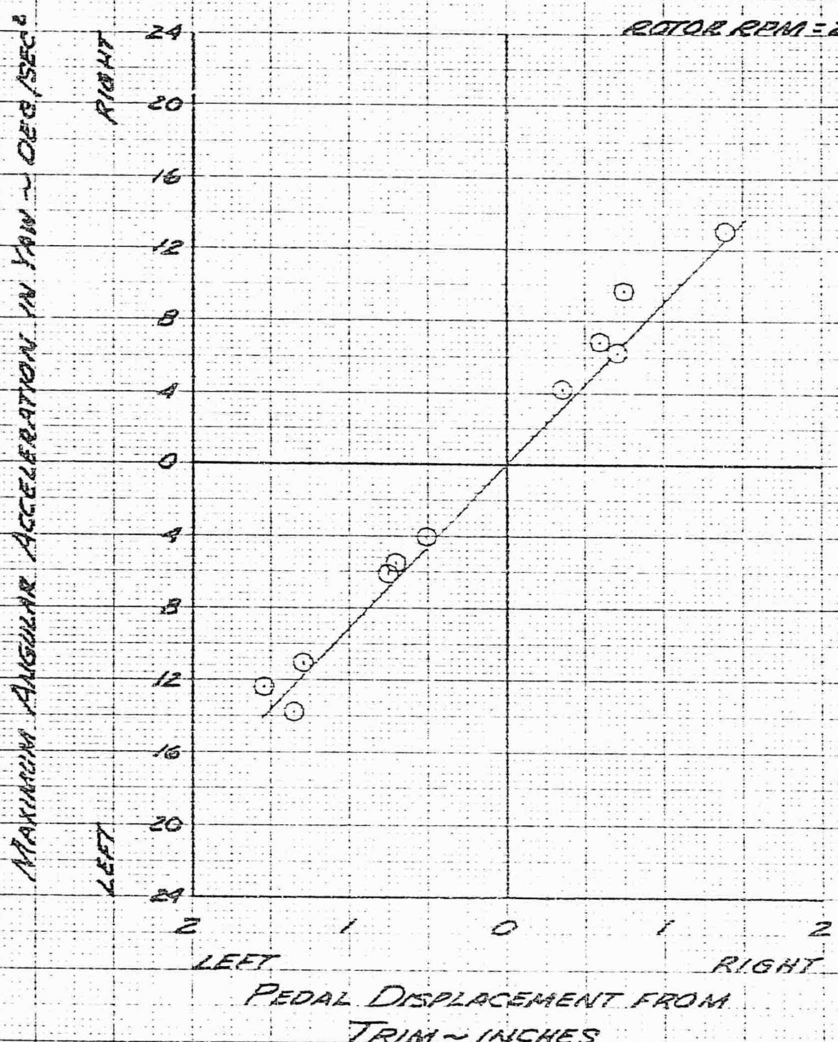
DIRECTIONAL CONTROL SENSITIVITY

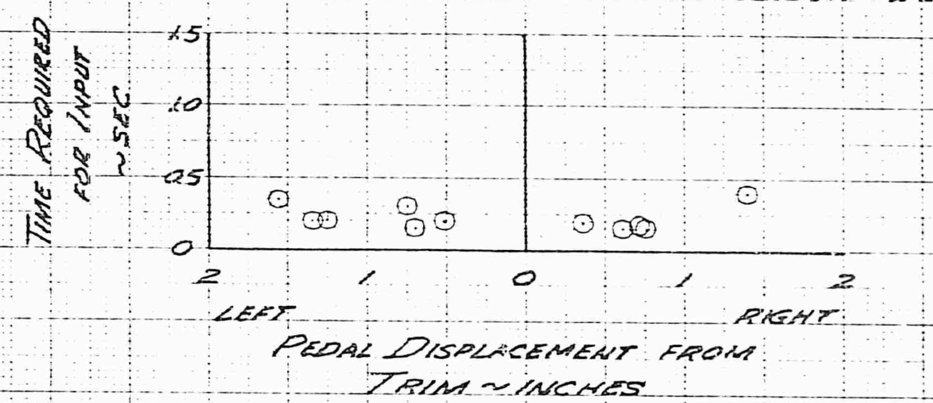
YHC-1A — S/N 58-5514

DIRECTIONAL AXIS OF SAS SYSTEM INOPERATIVE

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 14900 LBS AVG CG = 16.5 IN. FWD

ROTOR RPM = 258





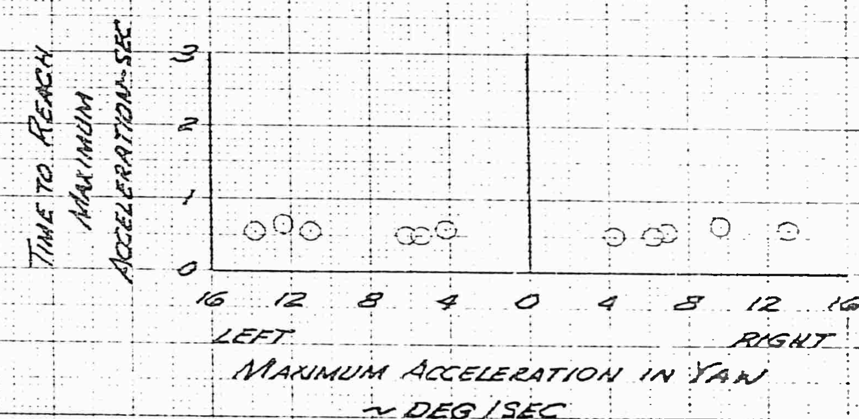


FIGURE No71

DIRECTIONAL CONTROL SENSITIVITY

YHC-1A S/N58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 32.5 KTS DENSITY ALT = 1000 FT AVG GW = 15150 LBS AVG CG = 19 IN. FWD

ROTOR RPM = 258

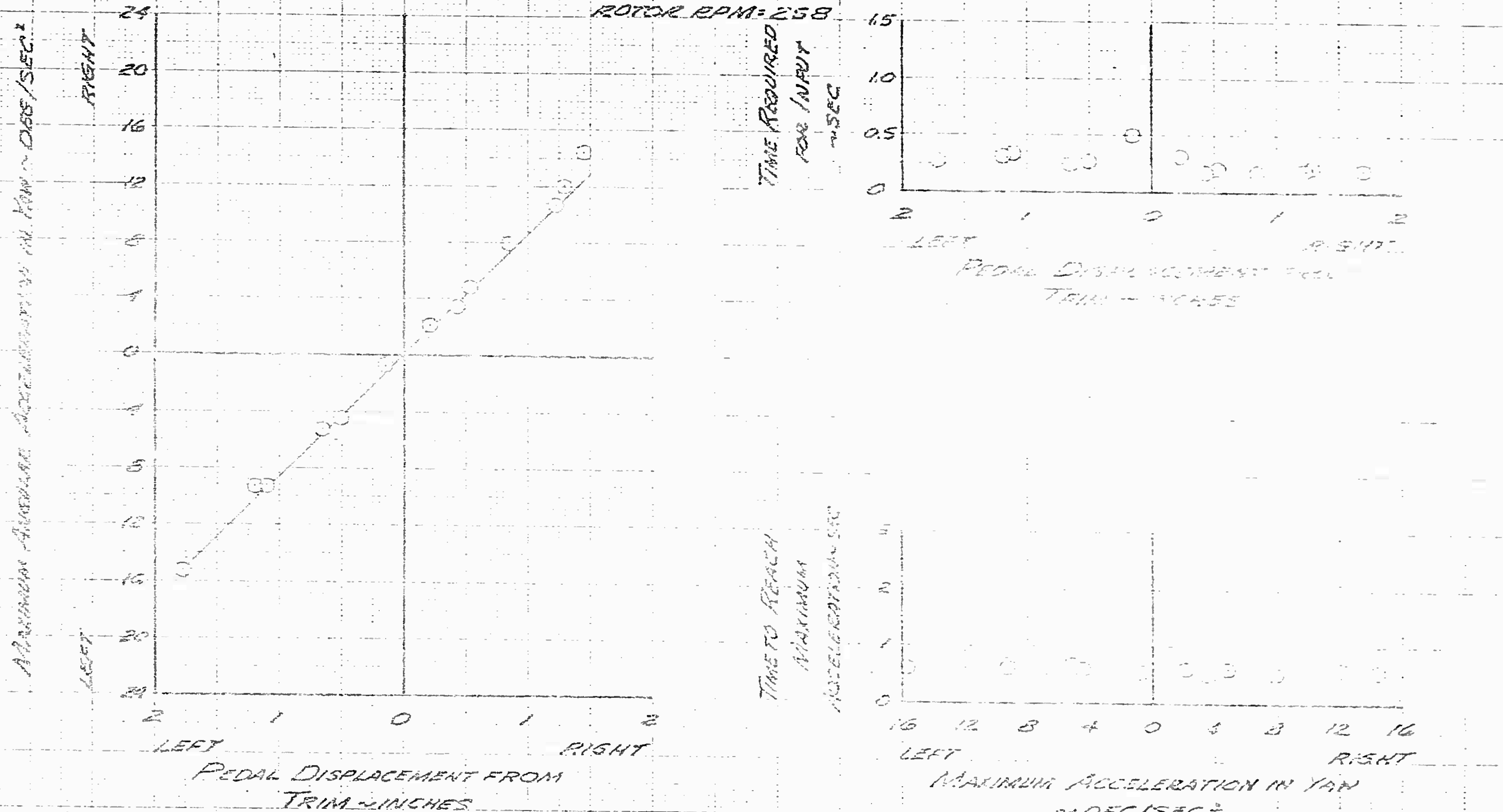


FIGURE No 72

DIRECTIONAL CONTROL SENSITIVITY

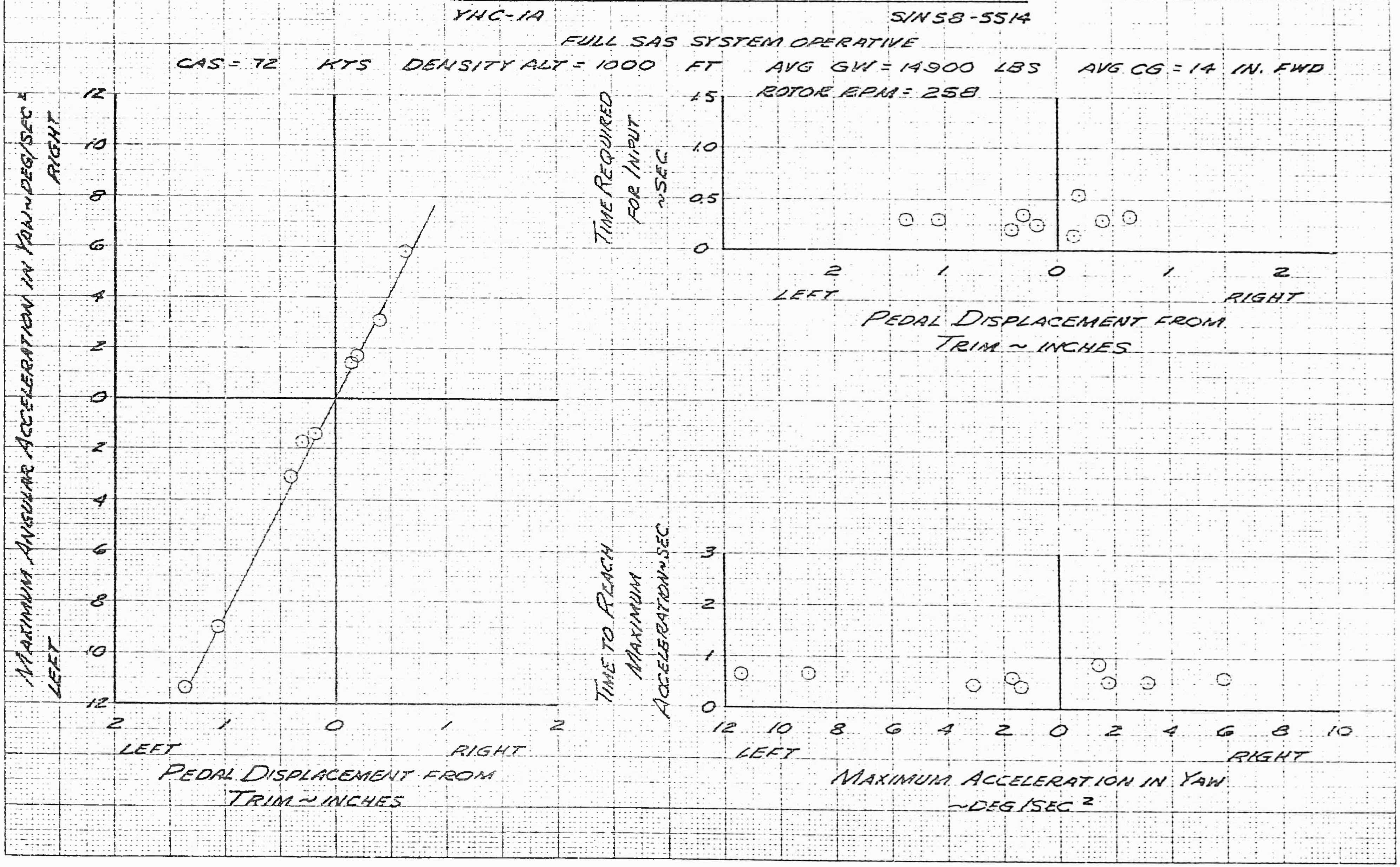


FIGURE No. 73

DIRECTIONAL CONTROL SENSITIVITY

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 109 KTS DENSITY ALT = 1000 FT AVG GW = 14600 LBS AVG CG = 19 IN. FWD

ROTOR RPM = 258

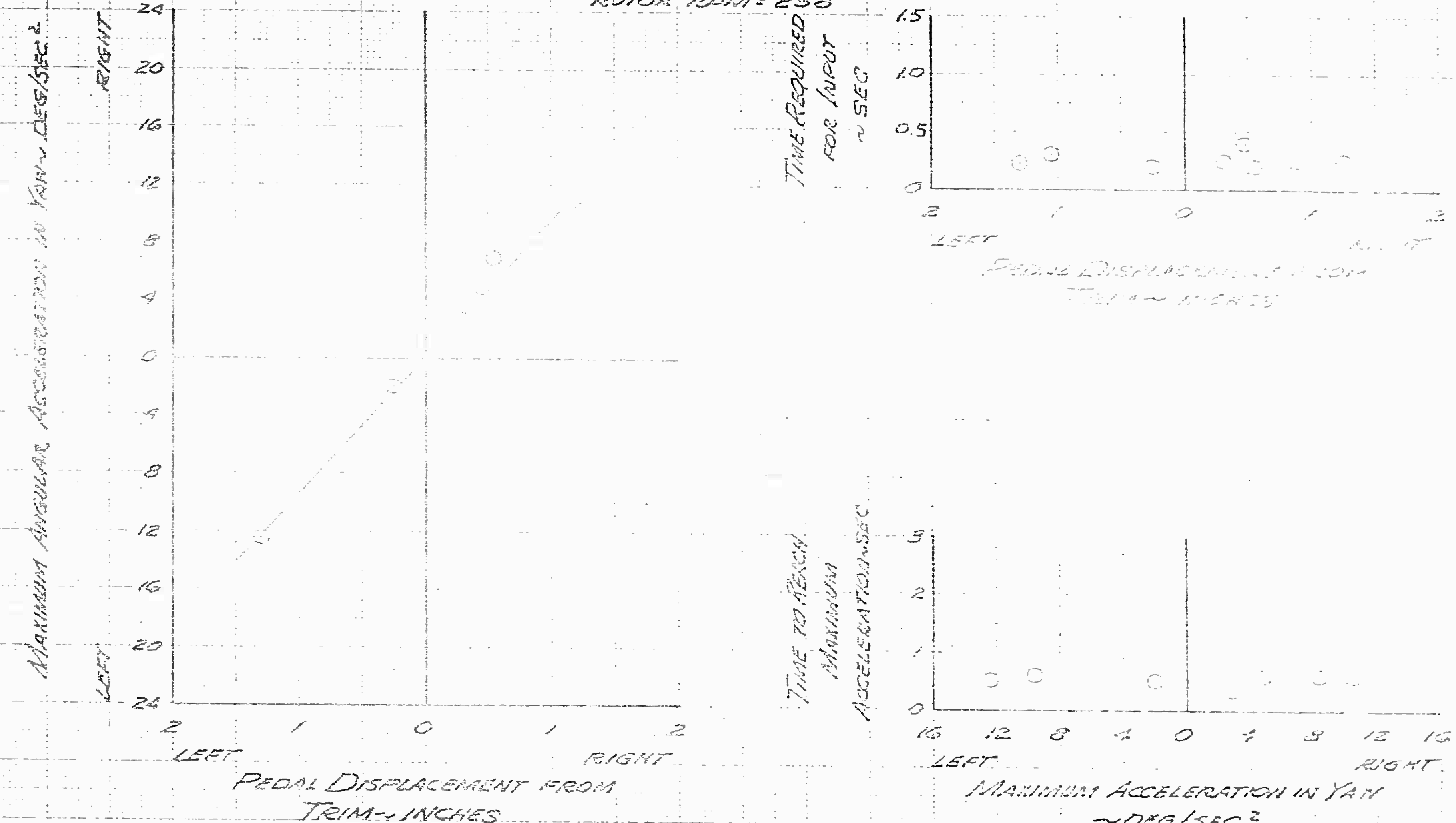


FIGURE NO 74

SUMMARY OF RESPONSE TO STEP INPUT

YHC-1A — S/N 58-5514

AVG GW = 15000 LBS — AVG CG = 20 FWD

Hd = 1000 FT — ROTOR RPM = 258

FULL SAS SYSTEM OPERATIVE

DERIVED FROM FIG NO 75 THROUGH 87

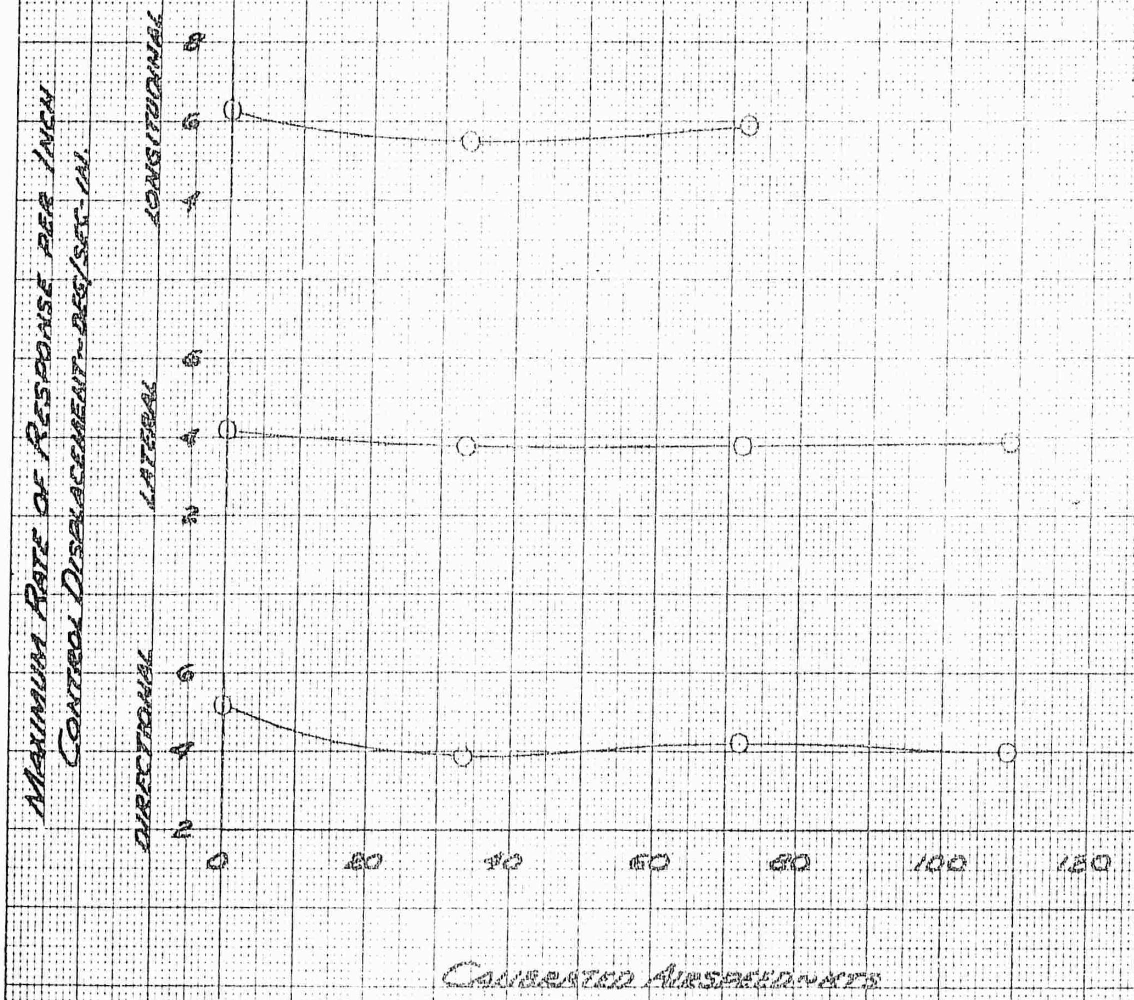


FIGURE NO. 75

RESPONSE TO LONGITUDINAL CONTROL INPUTS

YHC-1A S/N 58-5514

FULL SAS OPERATING

CAS = 0 KTS DENSITY ALT = 1000 FT AVG. GW = 15000 LBS AVG CG = 2.50 IN. FWD.

ROTOR RPM = 258

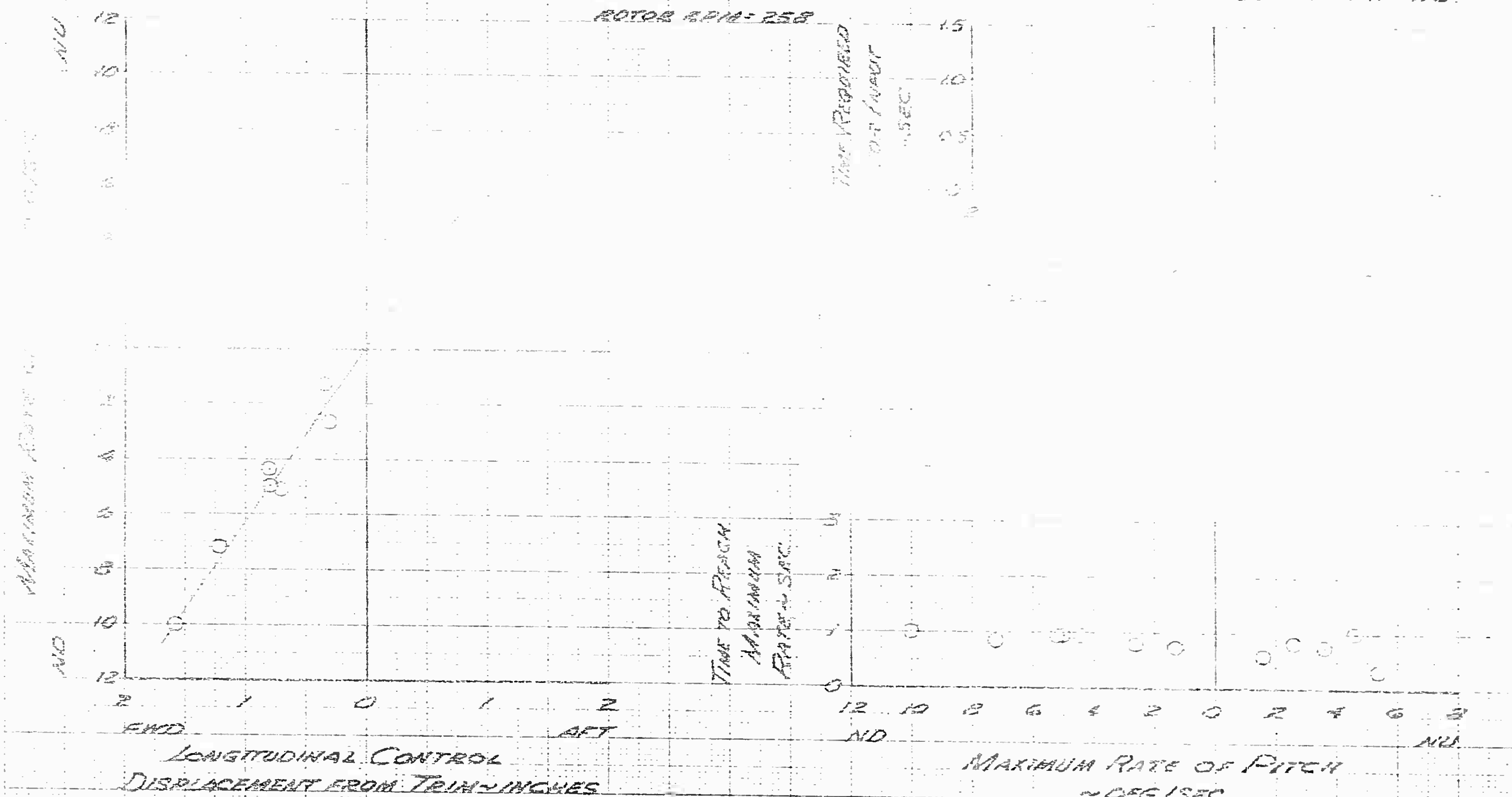


FIGURE No 76

RESPONSE TO LONGITUDINAL CONTROL INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 72 KTS DENSITY ALT = 1000 FT AVG GW = 15100 LBS AVG CG = 22.6 IN. FWD

ROTOR RPM = 258

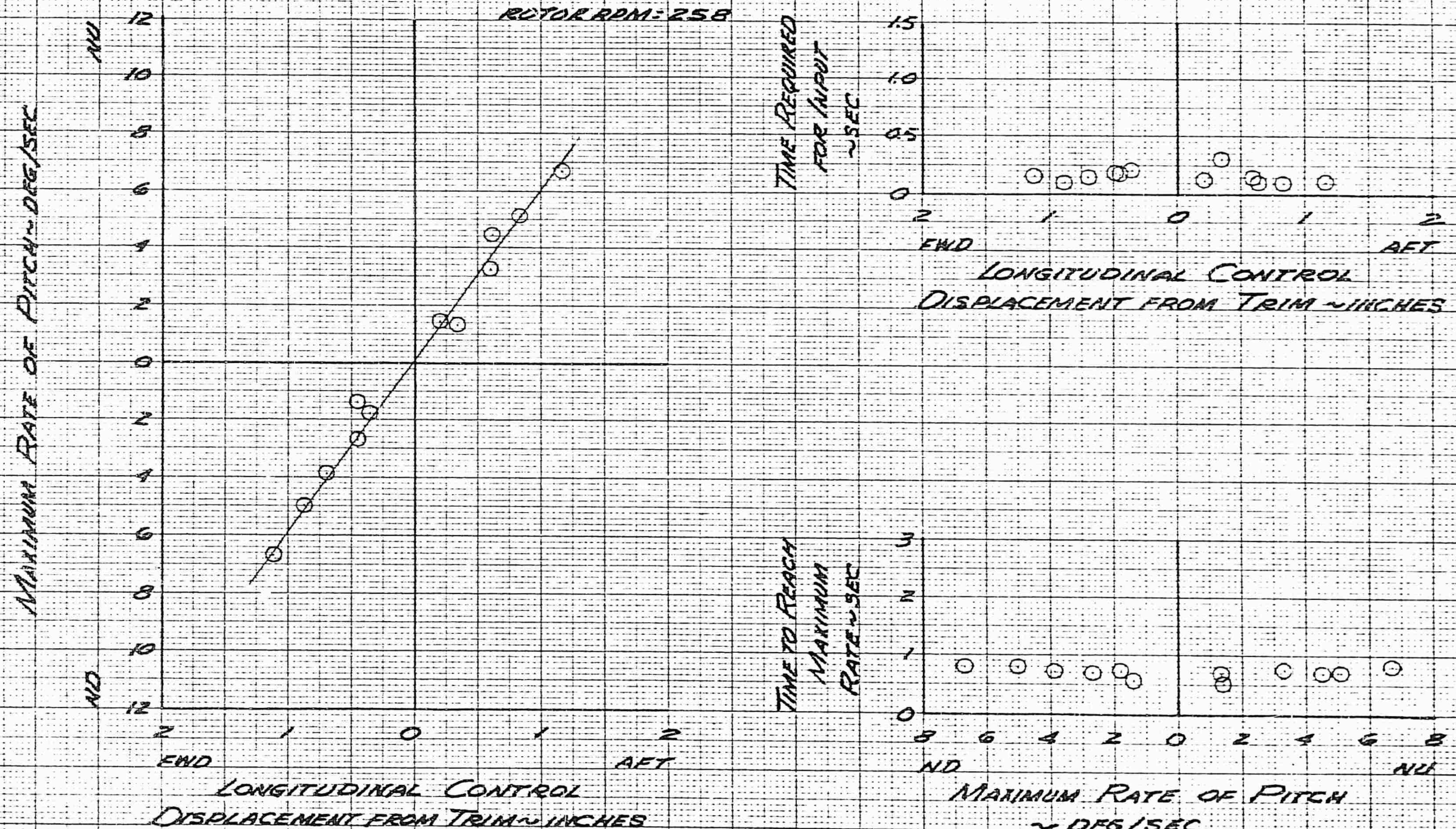


FIGURE No 77

RESPONSE TO LONGITUDINAL CONTROL INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS=33.5 KTS DENSITY ALT=1000 FT AVG GW=15250 LBS AVG CG=22.1 IN. FWD

ROTOR RPM=258

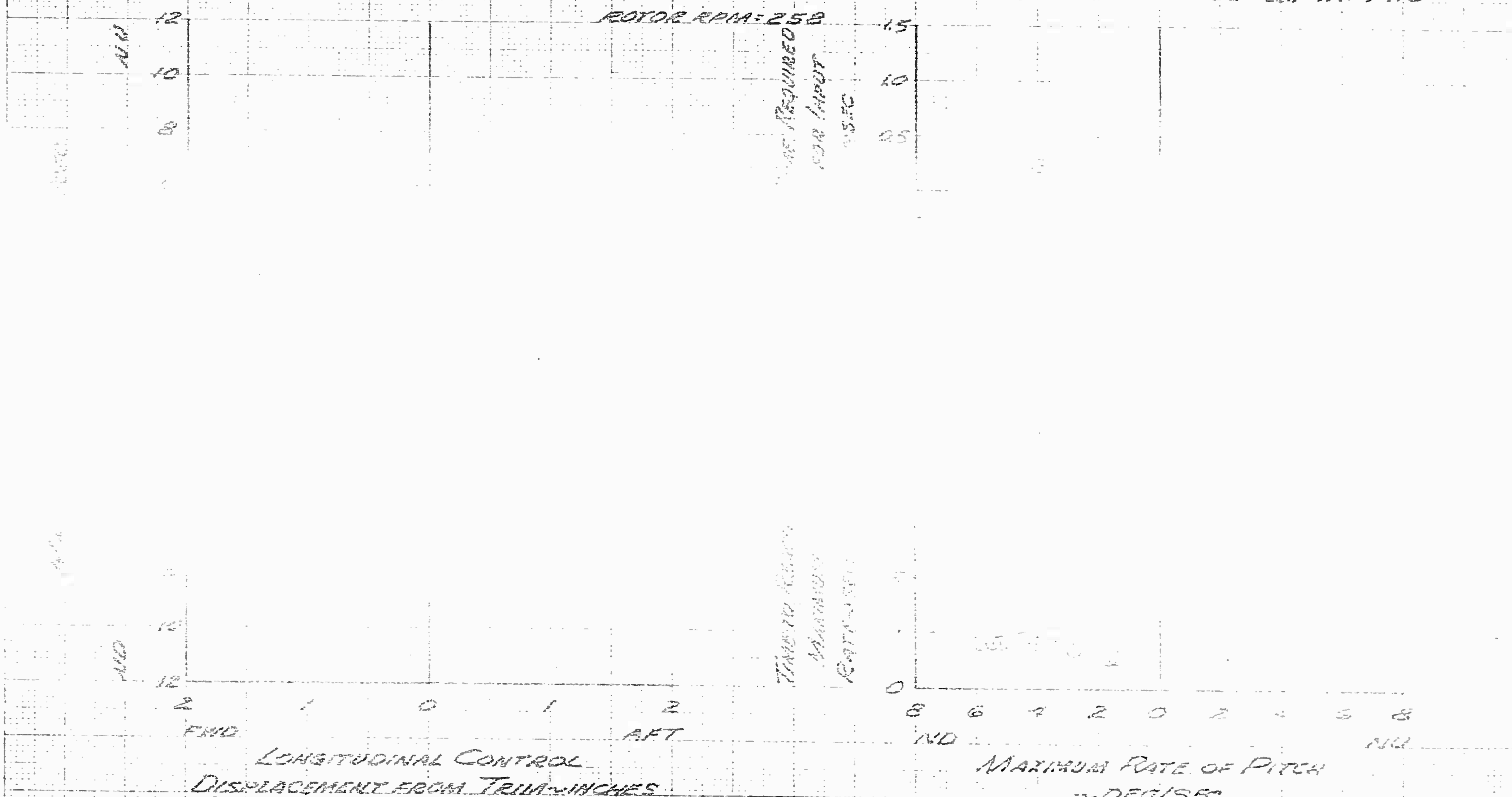


FIGURE No 78

RESPONSE TO LATERAL CYCLIC INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 0 KTS DENSITY ALT. = 1000 FT AVG GW = 15000 LBS AVG CG = 25.0 IN. FWD

ROTOR RPM = 258

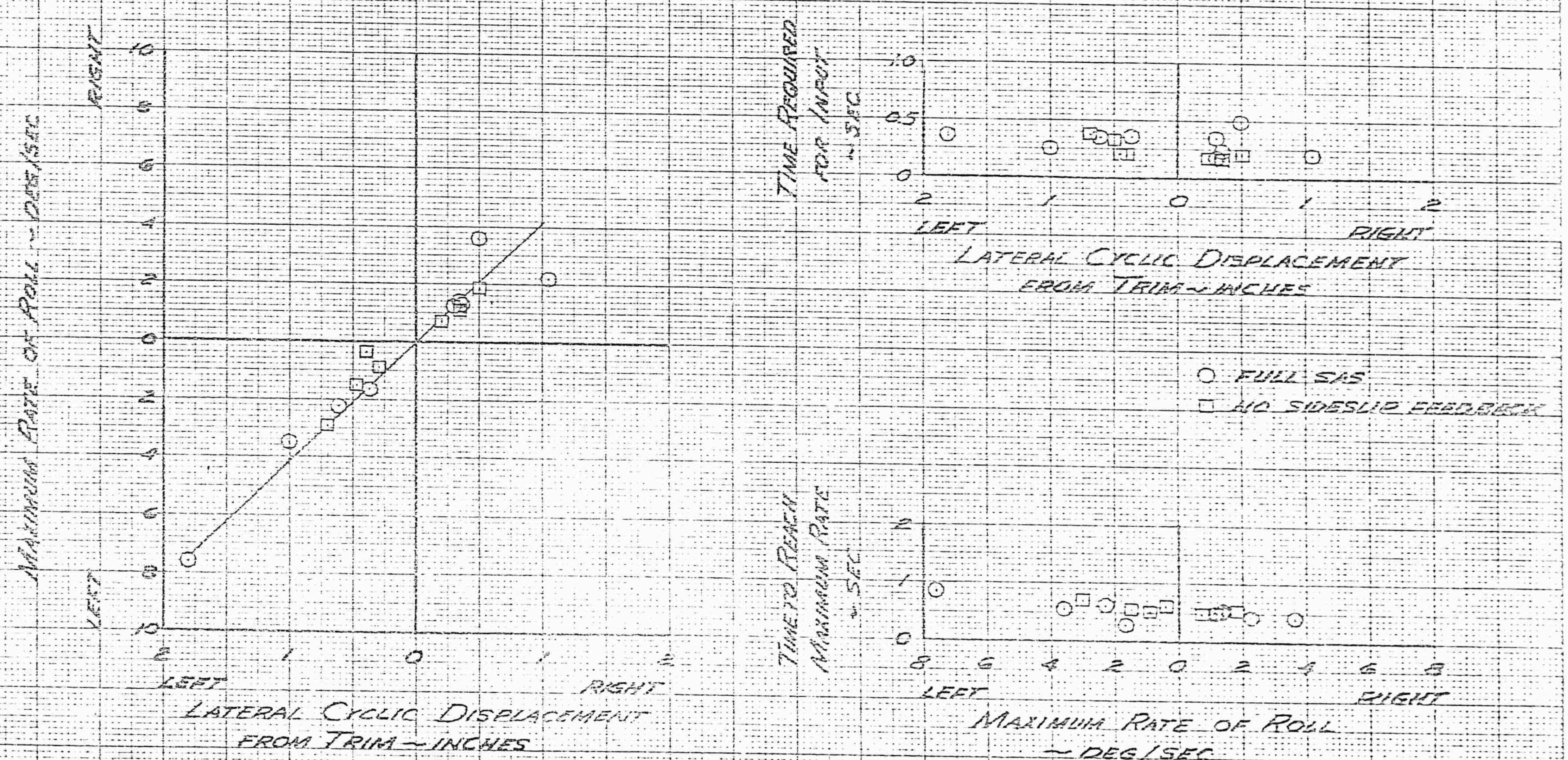


FIGURE No 79
RESPONSE TO LATERAL CYCLIC INPUTS
YHC-1A S/N 58-5514
SAS SYSTEM OPERATING AT HALF GAIN
CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 14500 LBS AVG CG = 25.6 IN. FWD
ROTOR RPM = 258

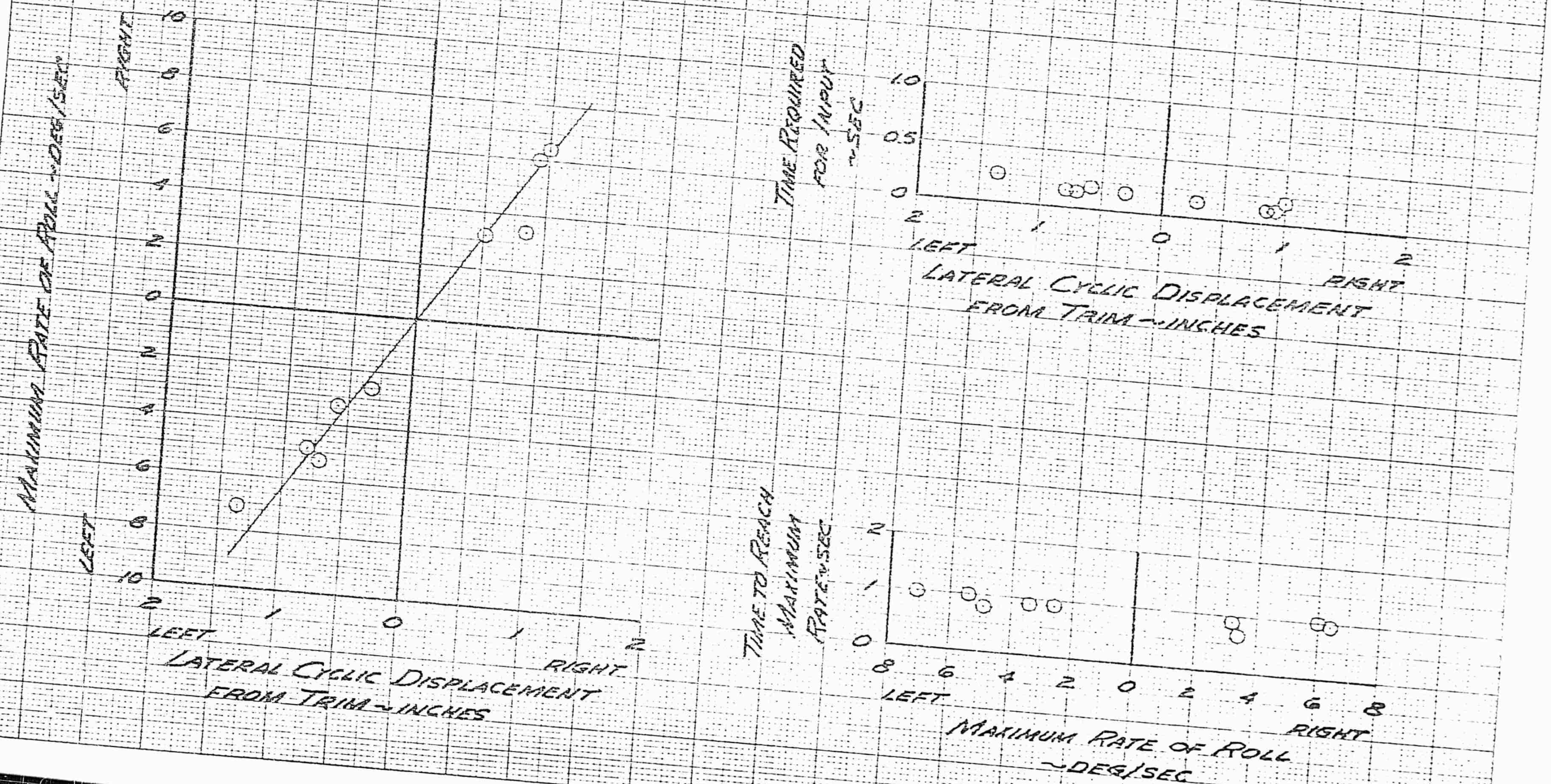


FIGURE No 80

RESPONSE TO LATERAL CYCLIC INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS=33.5 KTS DENSITY ALT=1000 FT AVG GW=15150 LBS AVG CG=19 IN FWD

ROTOR RPM=258

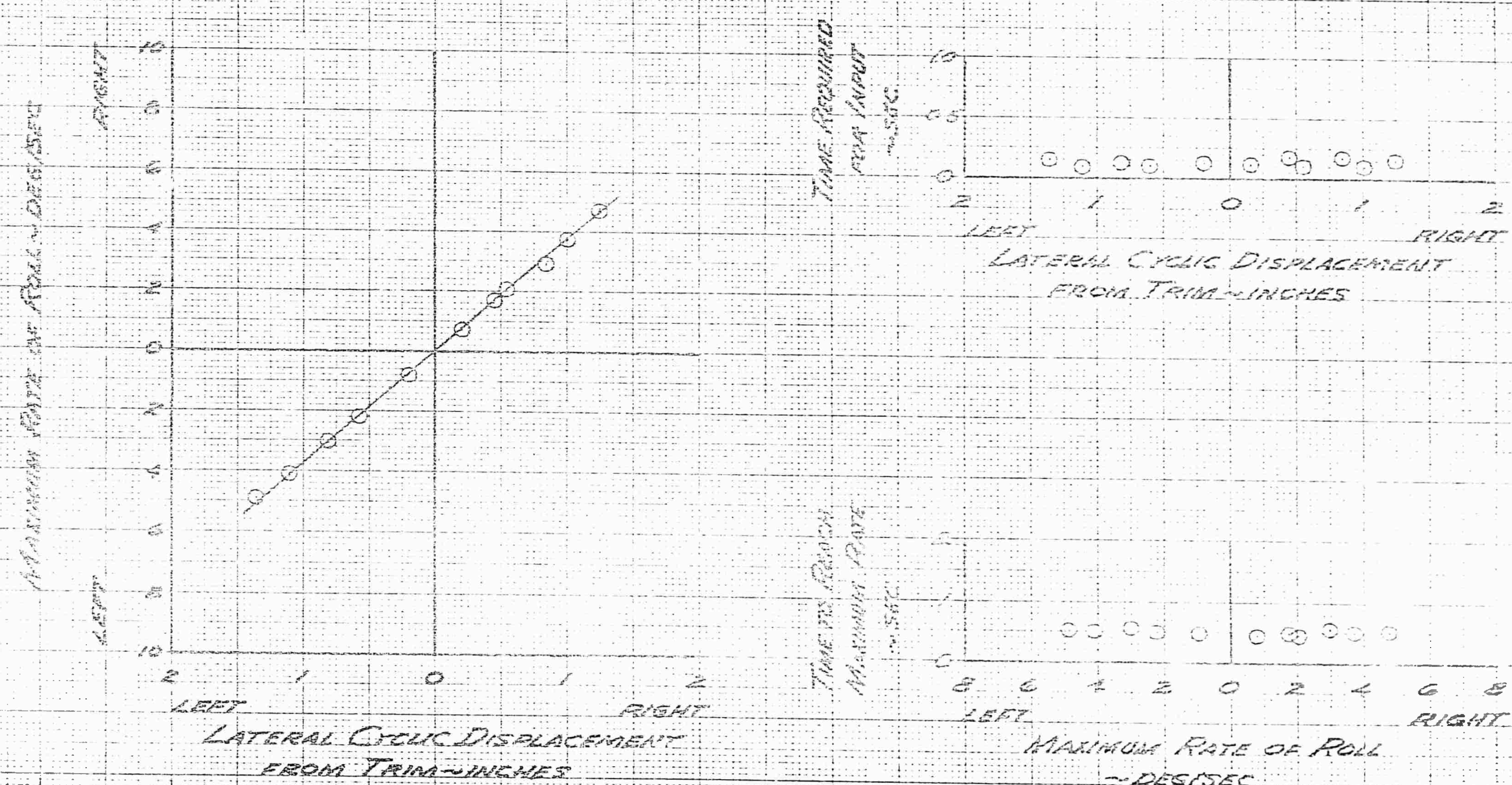


FIGURE No 81

RESPONSE TO LATERAL CYCLIC INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 72 KTS DENSITY ALT = 1000 FT AVG GW = 14950 LBS AVG CG = 19 IN. FWD

ROTOR RPM = 258

FIGURE NO 82

RESPONSE TO LATERAL CYCLIC INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 109 KTS DENSITY ALT = 1000 FT AVG GW = 14600 LBS AVG CG = 19 IN FWD

ROTOR RPM = 258

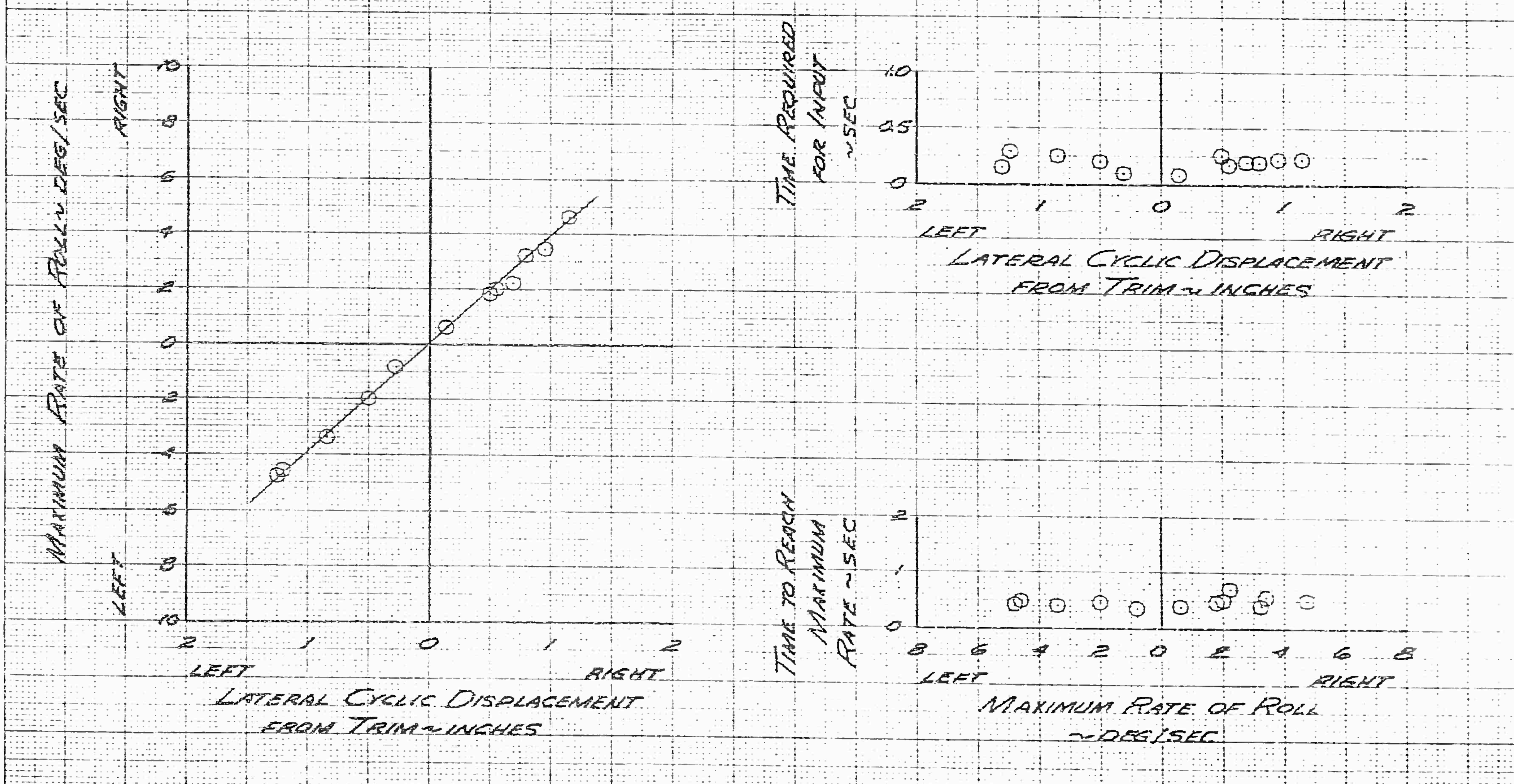


FIGURE No 83

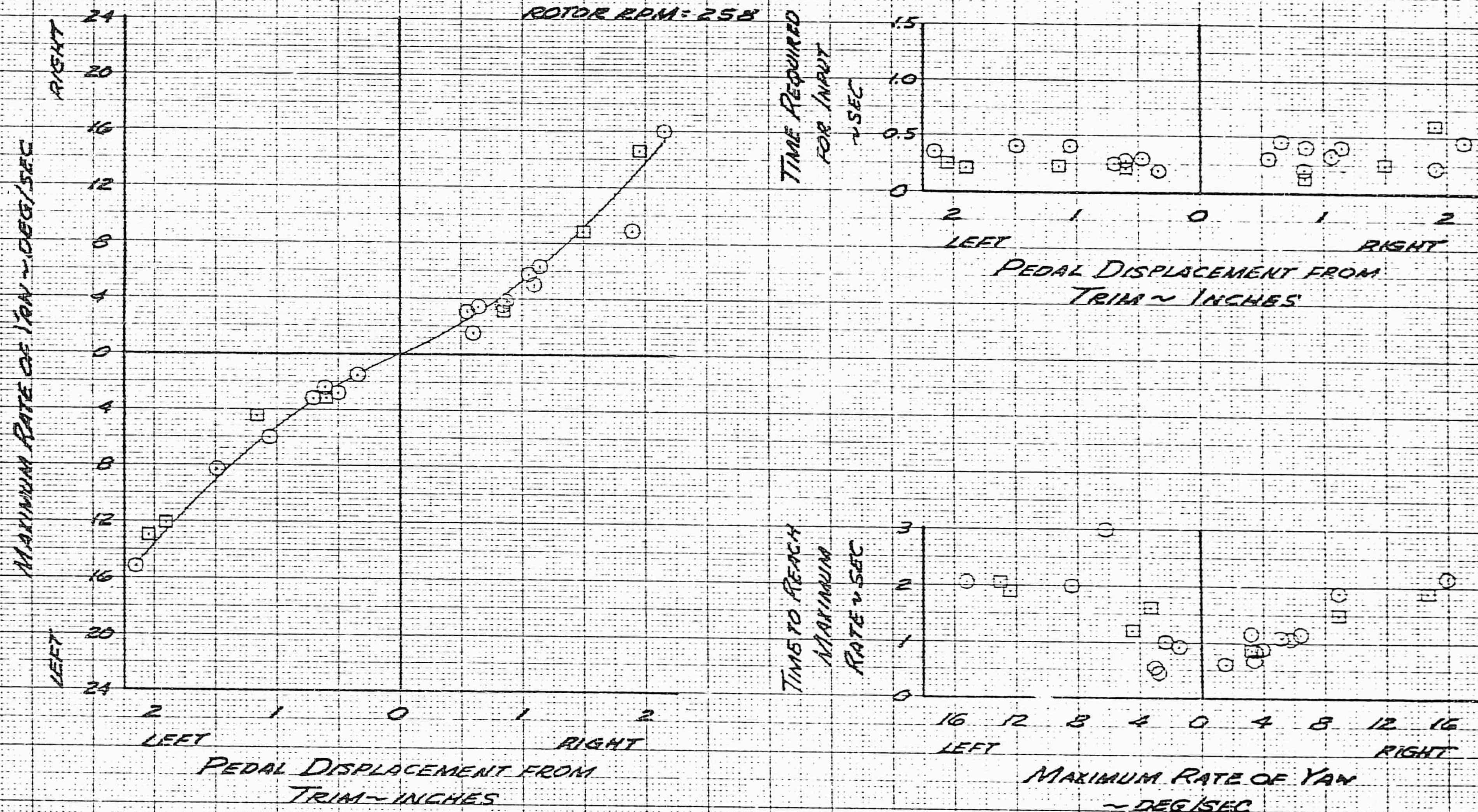
RESPONSE TO DIRECTIONAL CONTROL INPUTS

YHC-1A — SIN58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 0 KTS DENSITY ALT = 1000 FT AVG GW = 15050 LBS AVG CG = 19 IN. FWD

ROTOR RPM = 258



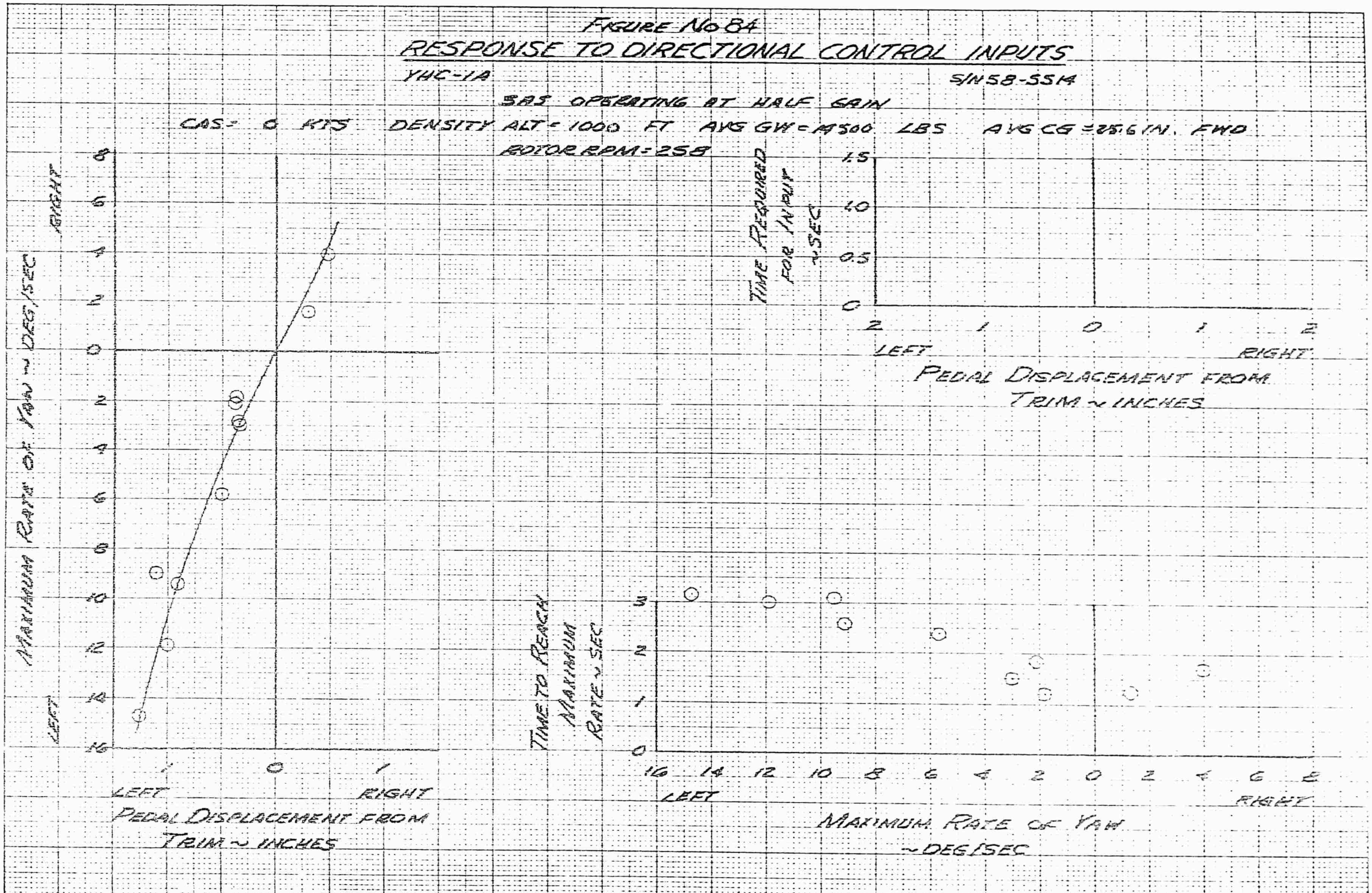


FIGURE No 84
RESPONSE TO DIRECTIONAL CONTROL INPUTS

FIGURE No 85

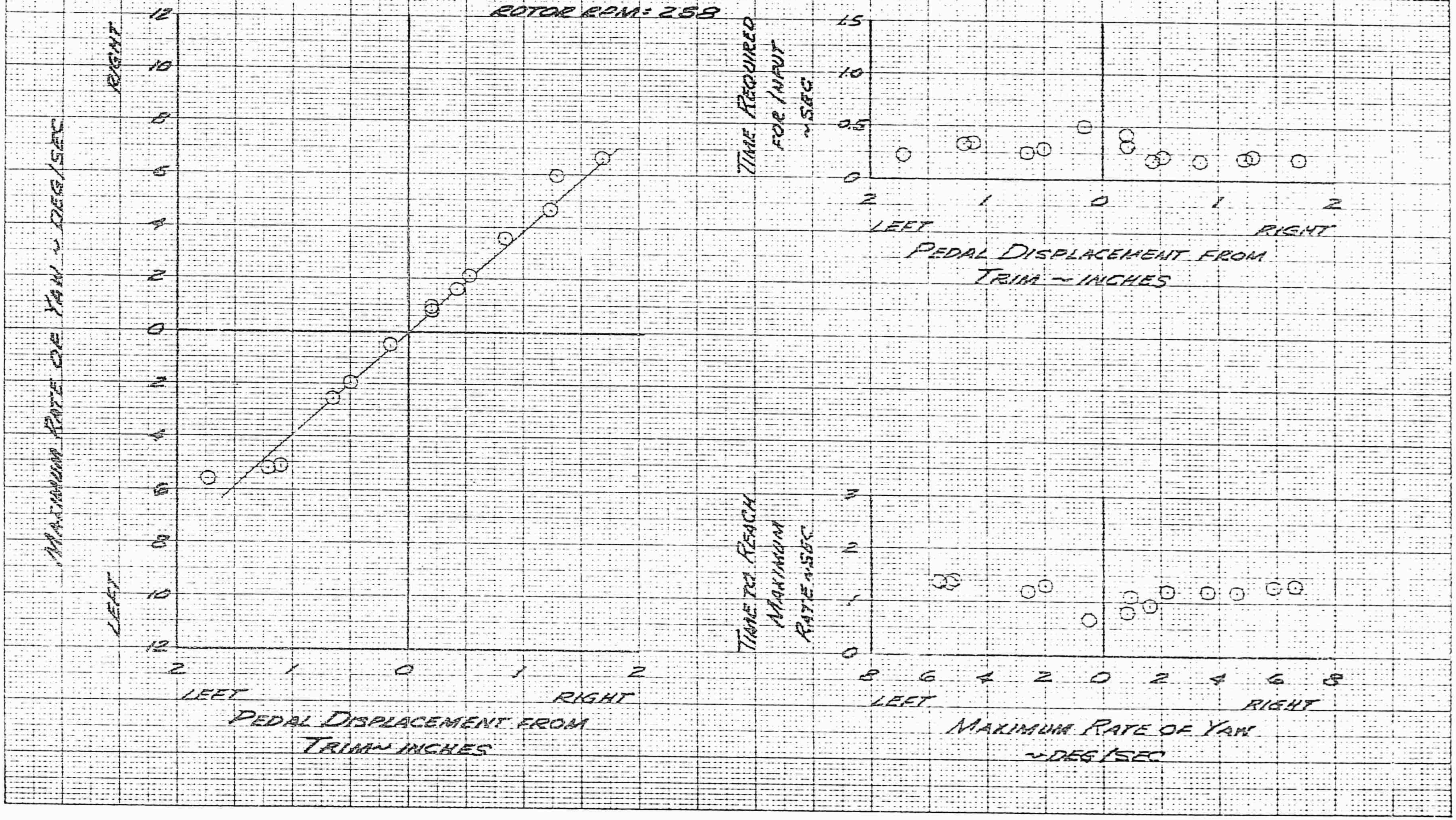
RESPONSE TO DIRECTIONAL CONTROL INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 33.5 KTS DENSITY ALT = 1000 FT AVG GW = 15150 LBS AVG CG = 1.9 IN. FWD

ROTOR RPM = 258



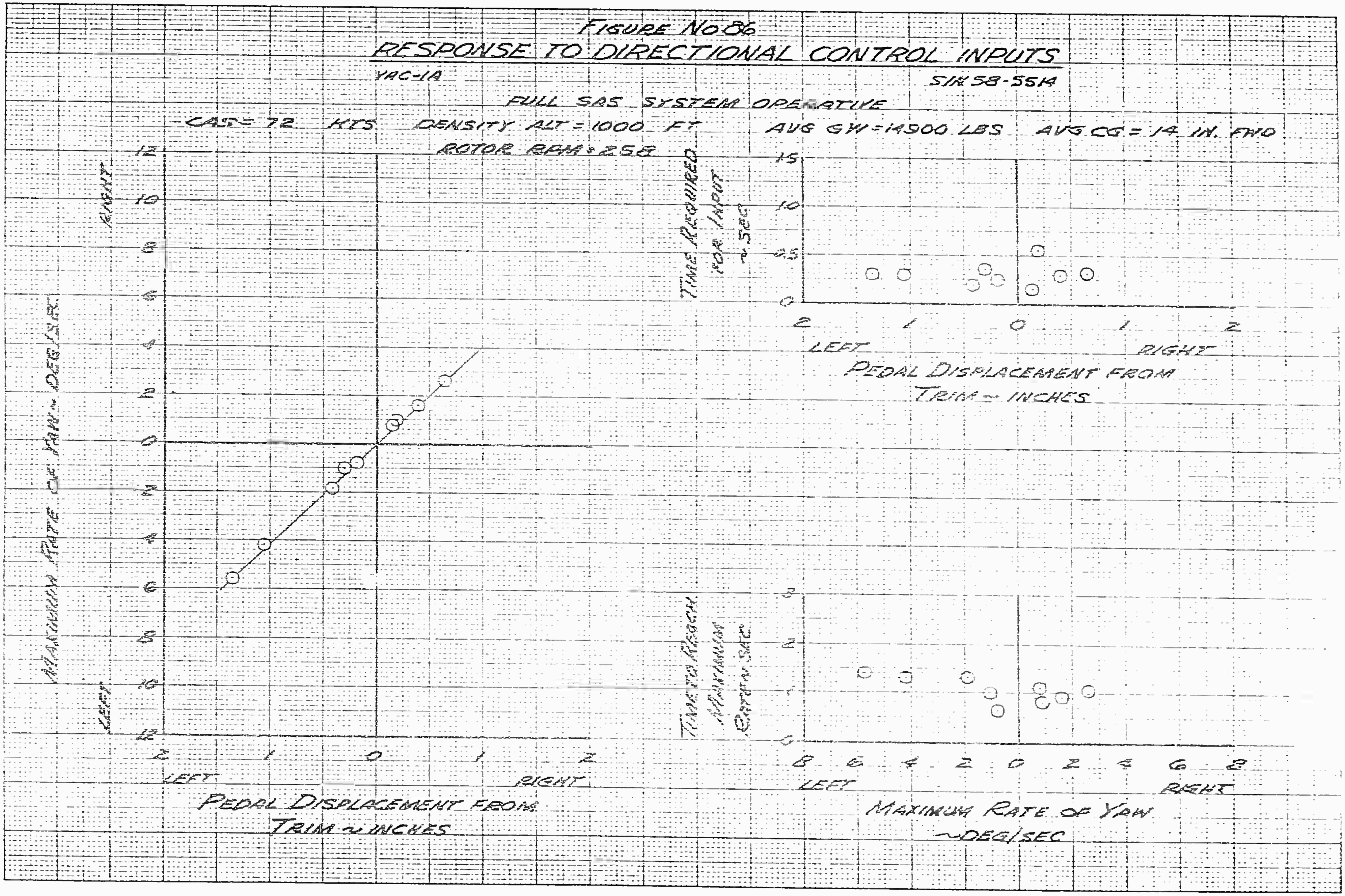


FIGURE No 86
RESPONSE TO DIRECTIONAL CONTROL INPUTS
YAC-1A
S/N 58-5514
FULL SAS SYSTEM OPERATIVE
CAS = 72 KTS
DENSITY ALT = 1000 FT
ROTOR RPM = 258
AVG GW = 14900 LBS
AVG CG = 14 IN. FWD
MAXIMUM RATE OF YAW ~ DEG/SEC
RIGHT
LEFT
12
10
8
6
4
2
0
2
4
6
8
10
12
2
1
0
1
2
LEFT
RIGHT
PEDAL DISPLACEMENT FROM
TRIM ~ INCHES
TIME REQUIRED
FOR INPUT
~ SEC
15
10
.5
0
2
1
0
1
2
LEFT
RIGHT
PEDAL DISPLACEMENT FROM
TRIM ~ INCHES
TIME TO REACH
MAXIMUM
RATE ~ SEC
3
2
1
0
8
6
4
2
0
2
4
6
8
LEFT
RIGHT
MAXIMUM RATE OF YAW
~DEG/SEC

FIGURE No 87

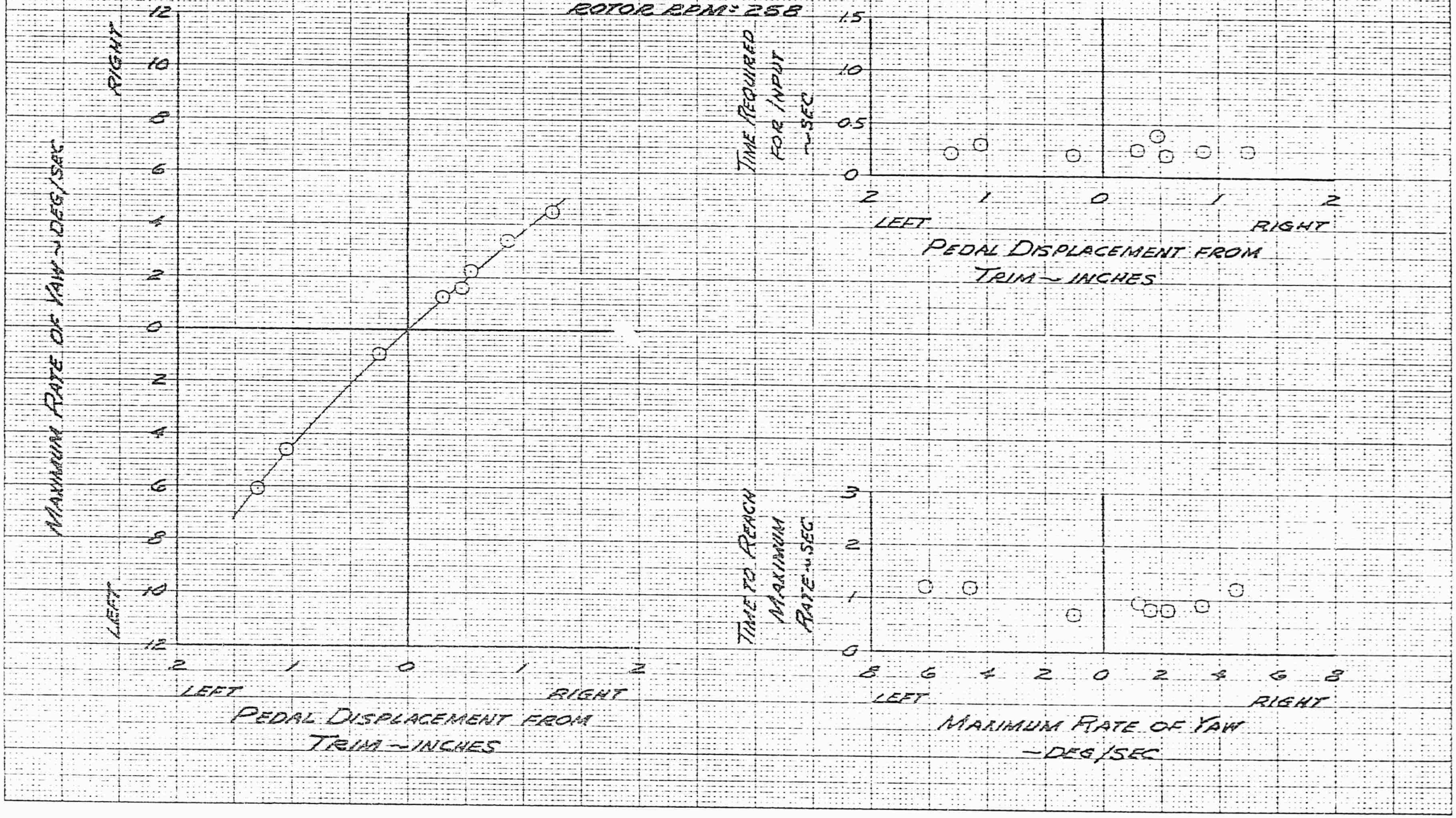
RESPONSE TO DIRECTIONAL CONTROL INPUTS

YHC-1A S/N 58-5514

FULL SAS SYSTEM OPERATIVE

CAS = 109 KTS DENSITY ALT = 1000 FT AVG GW = 14600 LBS AVG CG = 19 IN. FWD

ROTOR RPM = 258



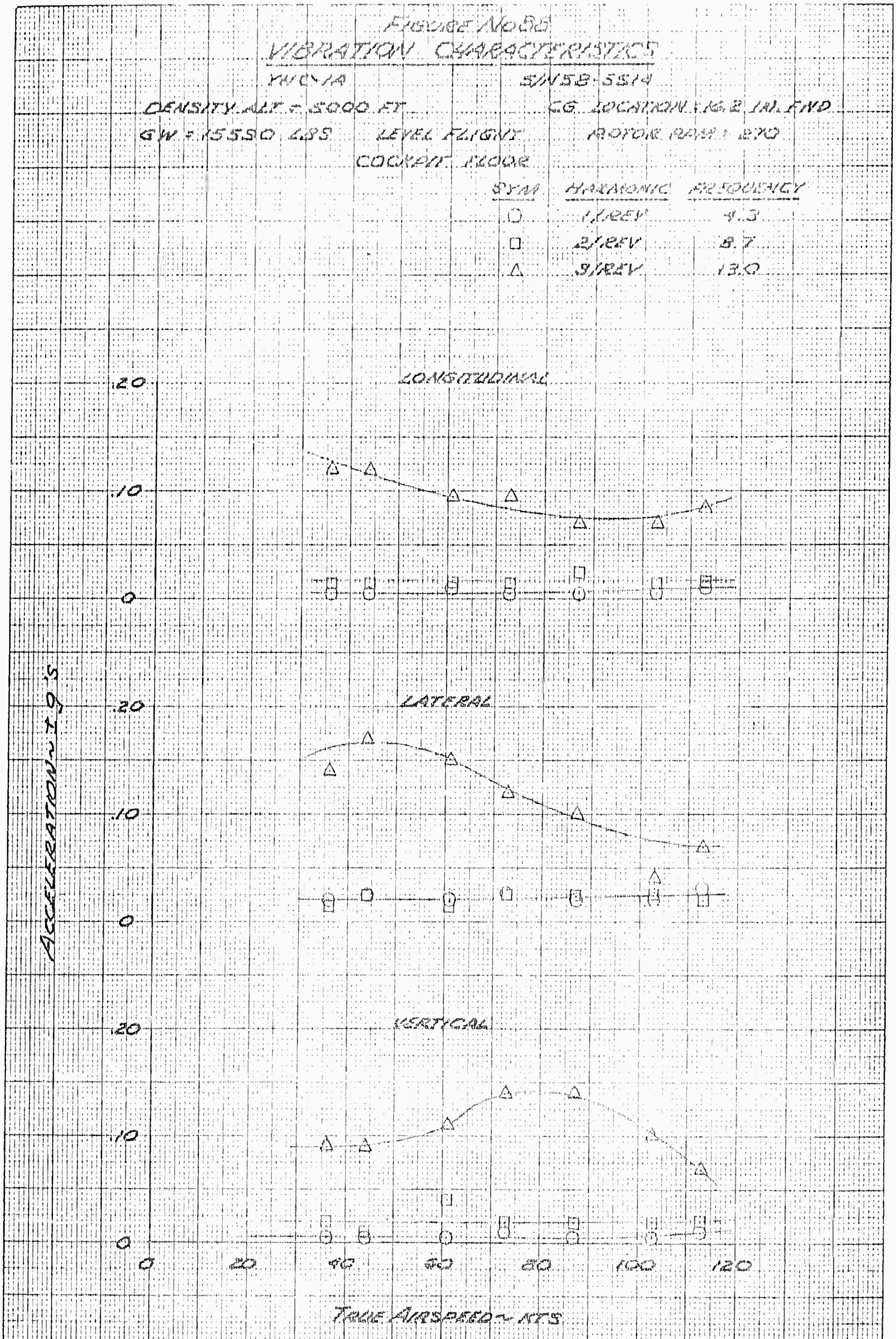

VIBRATION CHARACTERISTICS
YHC-1A
S/N 58-5514
DENSITY ALT = 5000 FT
GW = 15550 LBS
LEVEL FLIGHT
ROTOR RPM = 270
COCKPIT FLOOR
SYM
HARMONIC
FREQUENCY
1/REV
4.3
2/REV
8.7
3/REV
13.0
LONGITUDINAL
LATERAL
VERTICAL
ACCELERATION ~ ± g's
.20
.10
0
20
40
60
80
100
120
TRUE AIRSPEED ~ KTS

FIGURE NO 89

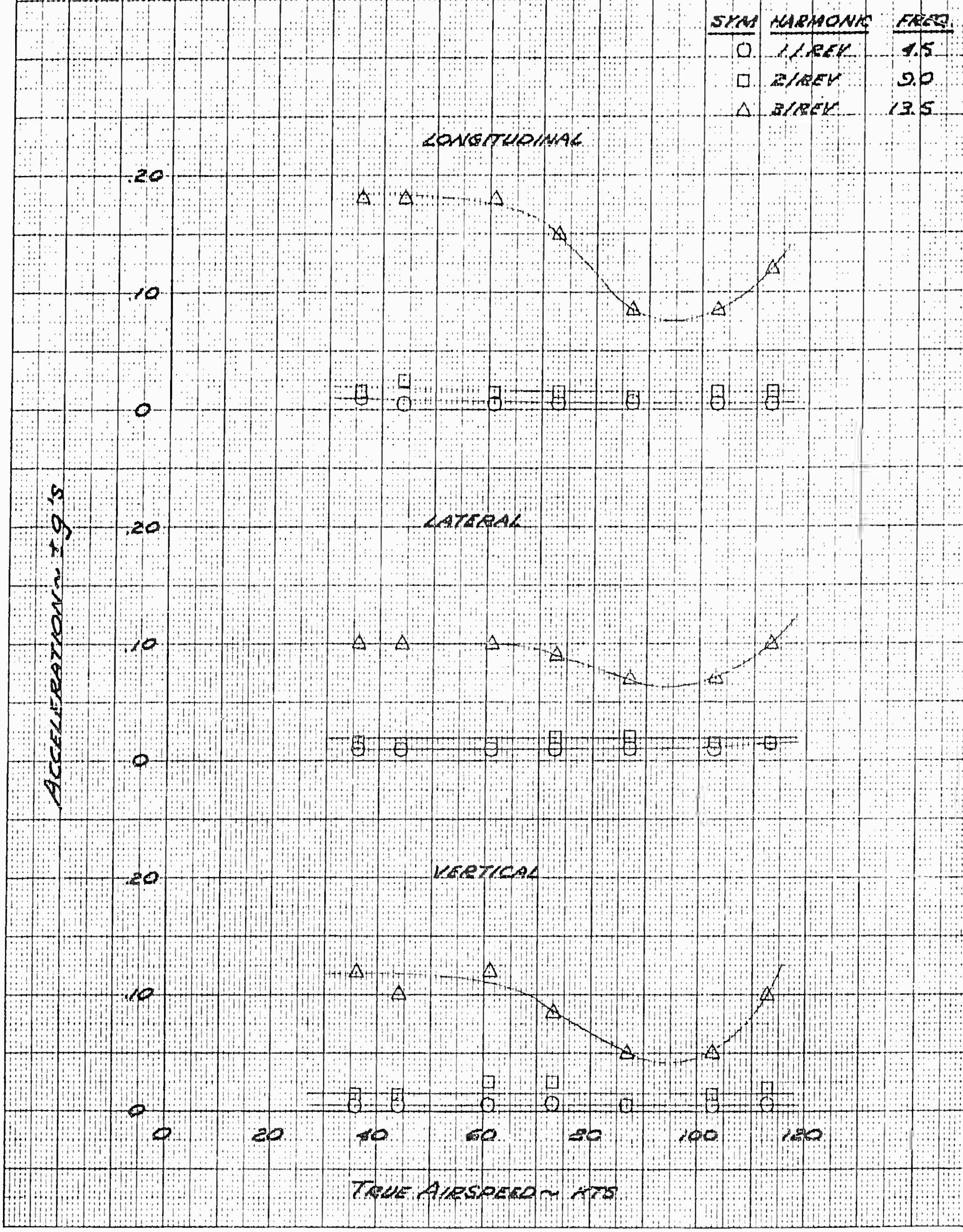
VIBRATION CHARACTERISTICS

YHC-1A S/N 58-5510

DENSITY ALT = 5000 FT CG LOCATION = 16.2 IN FWD

GW = 15550 LEVEL FLIGHT ROTOR RPM = 270

CARGO COMPARTMENT FLOOR AT AIRCRAFT CG



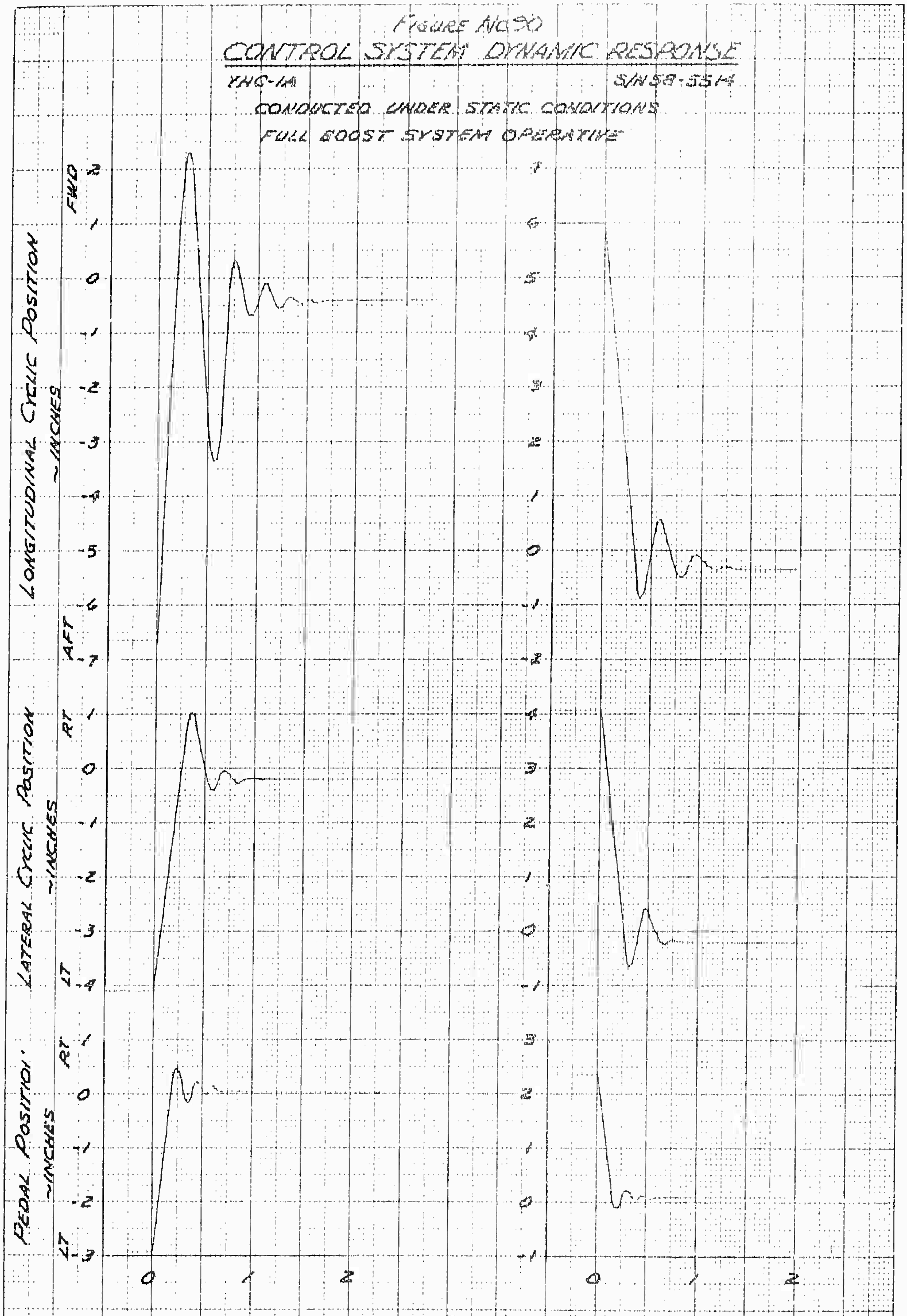


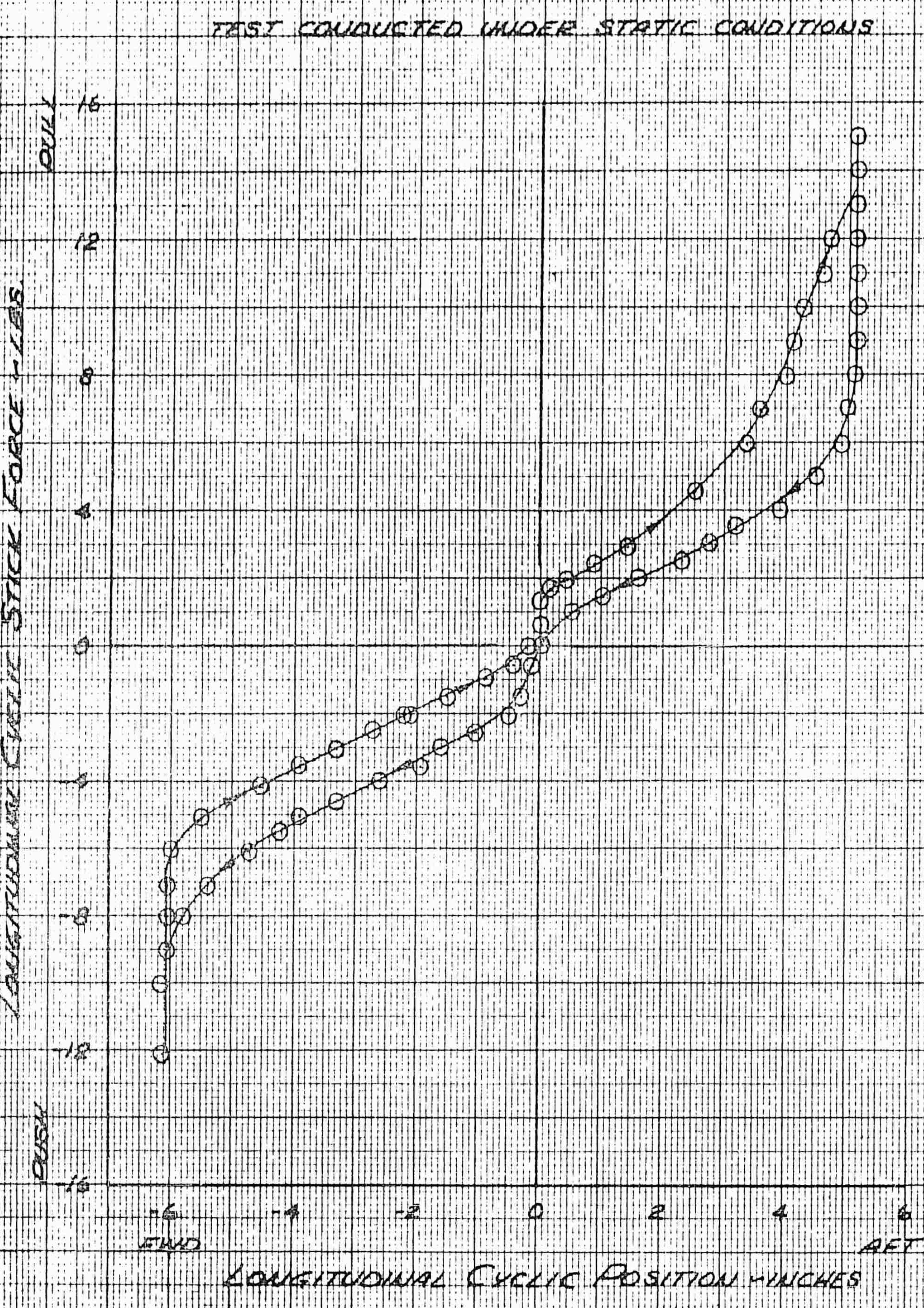
Figure No 90
CONTROL SYSTEM DYNAMIC RESPONSE
YHC-1A
S/N 58-5514
CONDUCTED UNDER STATIC CONDITIONS
FULL BOOST SYSTEM OPERATIVE
LONGITUDINAL CYCLIC POSITION ~INCHES
FWD
AFT
LATERAL CYCLIC POSITION ~INCHES
RT
LT
PEDAL POSITION ~INCHES
RT
LT
TIME~SECONDS

Figure No 51

LONGITUDINAL CONTROL SYSTEM FRICTION

YHC-1A S/N 58-5514

CONTROL BOOST ON



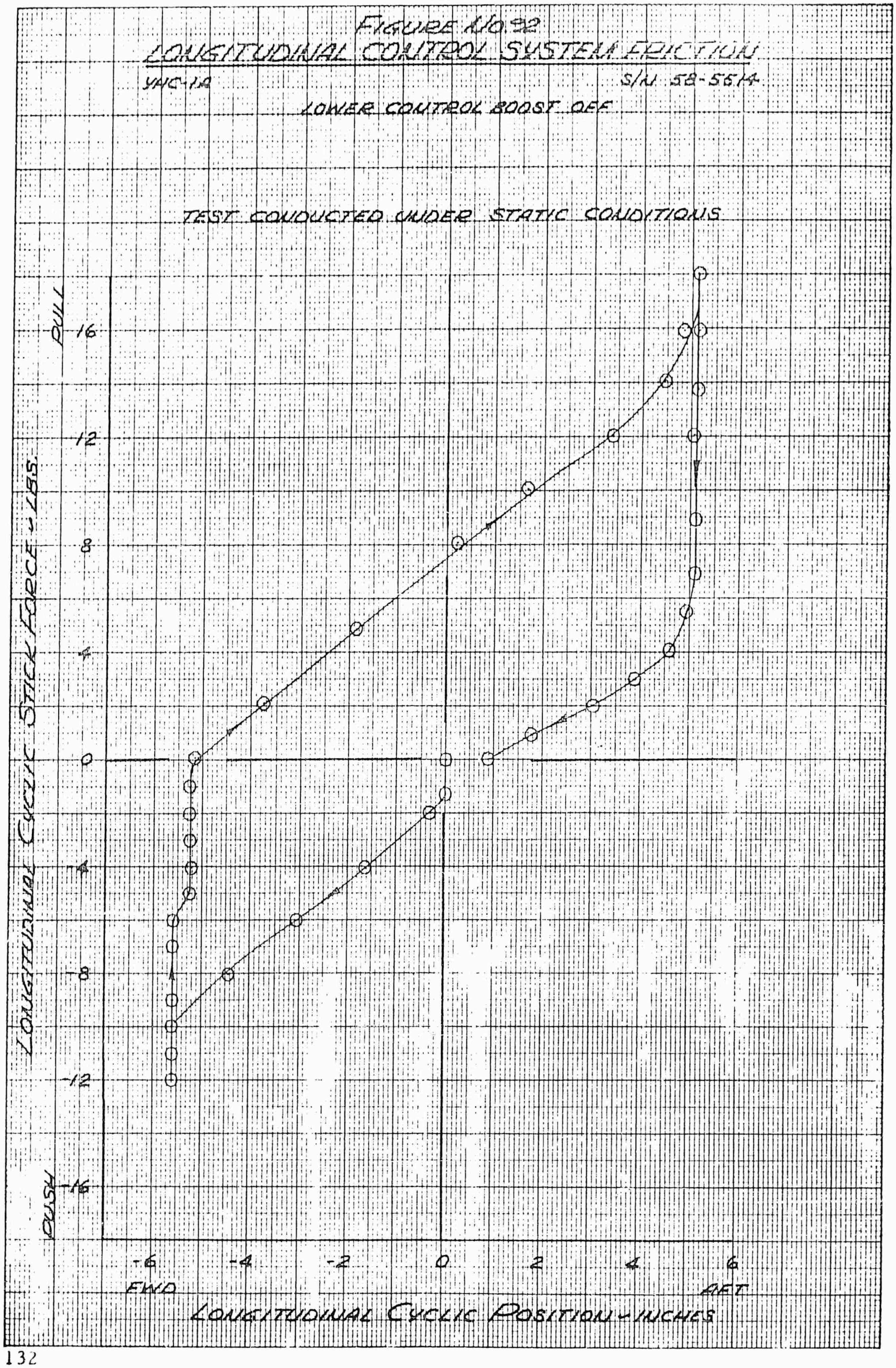


FIGURE NO. 92
LONGITUDINAL CONTROL SYSTEM FRICTION
YHC-1A
S/N 58-5514
LOWER CONTROL BOOST OFF
TEST CONDUCTED UNDER STATIC CONDITIONS
PULL
PUSH
16
12
8
4
0
-4
-8
-12
-16
LONGITUDINAL CYCLIC STICK FORCE - LBS.
-6
-4
-2
0
2
4
6
FWD
AFT
LONGITUDINAL CYCLIC POSITION - INCHES

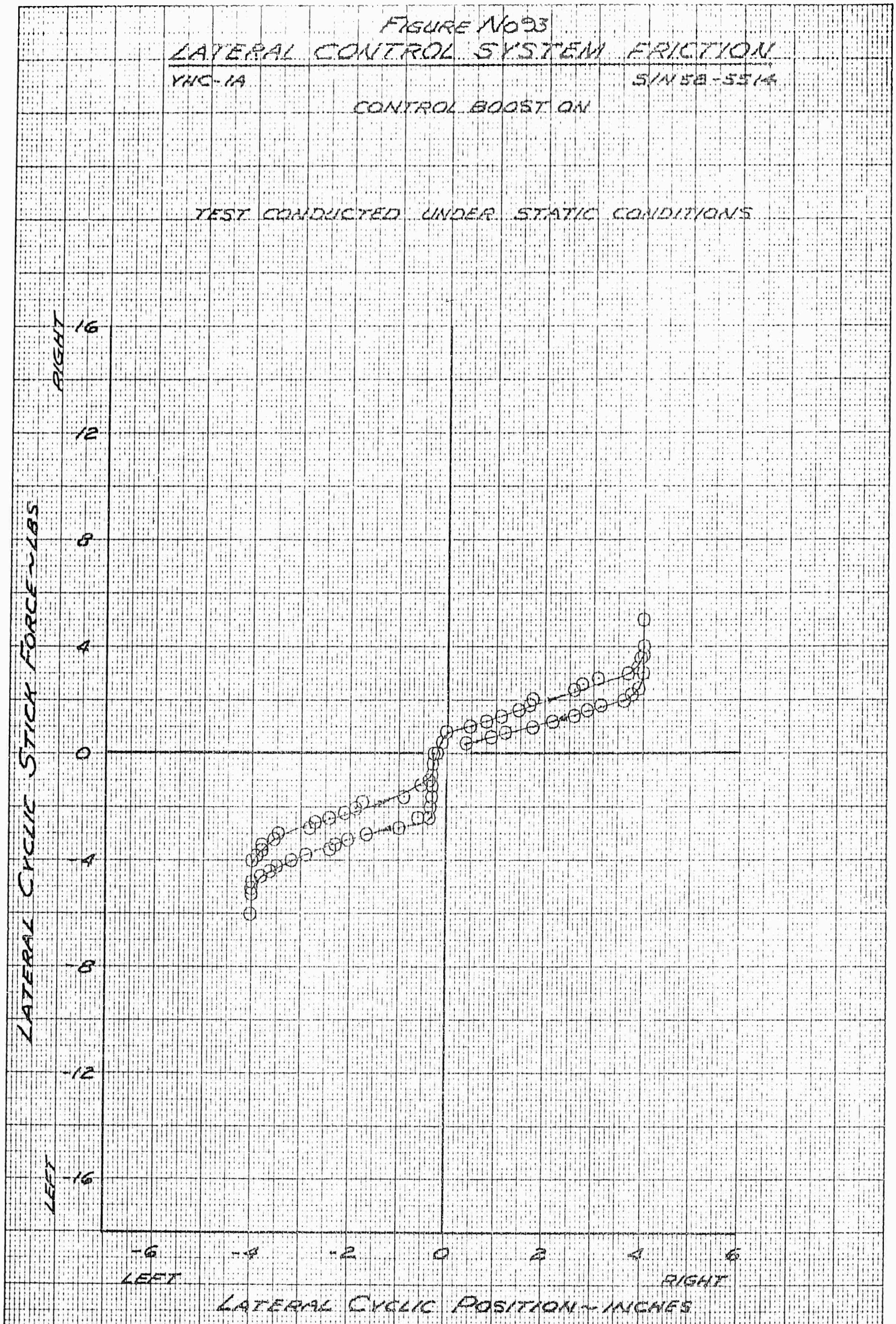


FIGURE No 93
LATERAL CONTROL SYSTEM FRICTION
YHC-1A
S/N 58-5514
CONTROL BOOST ON
TEST CONDUCTED UNDER STATIC CONDITIONS
LATERAL CYCLIC STICK FORCE ~ LBS
RIGHT
LEFT
16
12
8
4
0
-4
-8
-12
-16
-6
-4
-2
0
2
4
6
LEFT
RIGHT
LATERAL CYCLIC POSITION ~ INCHES

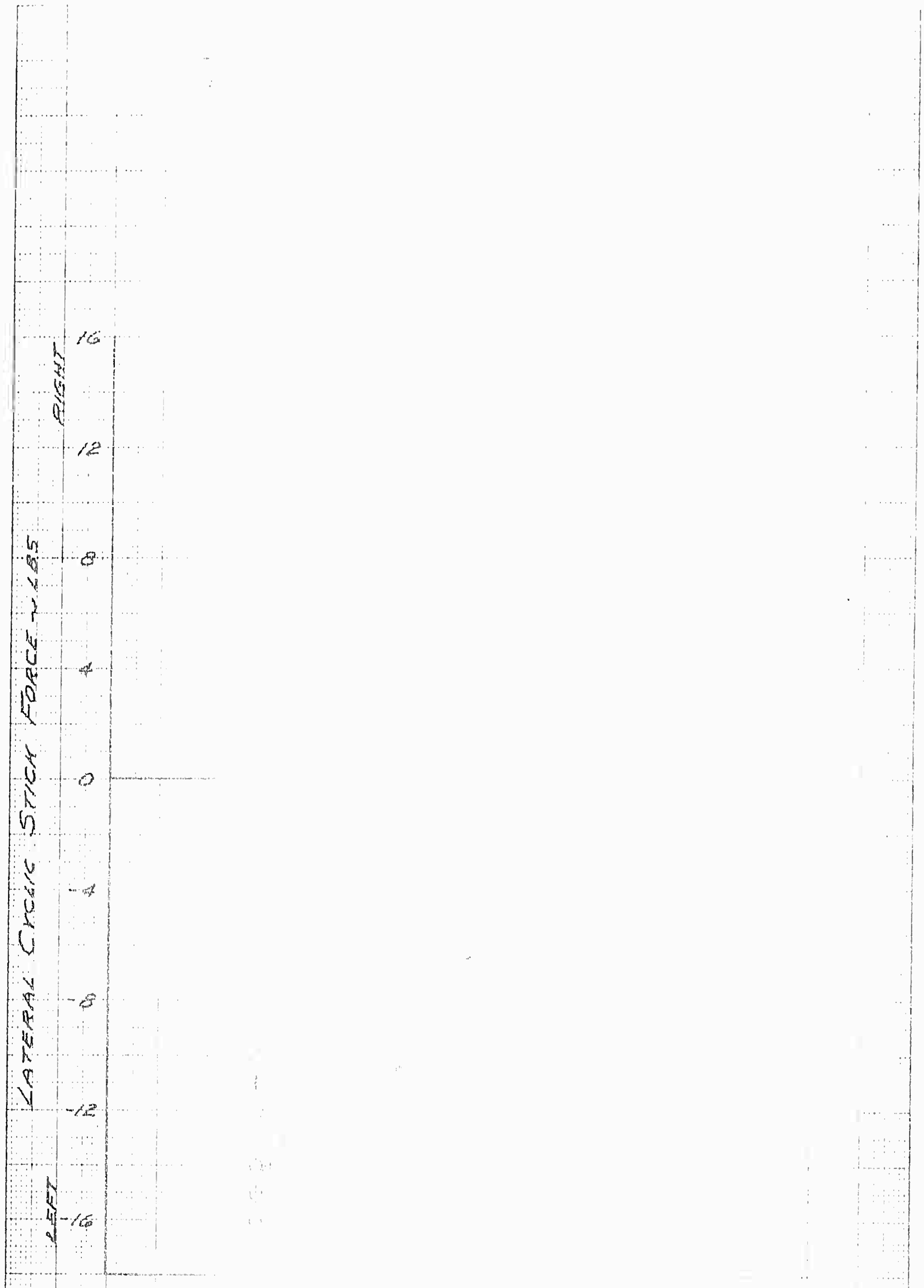

LATERAL CYCLIC STICK FORCE ~ LBS
RIGHT
LEFT
16
12
8
4
0
-4
-8
-12
-16

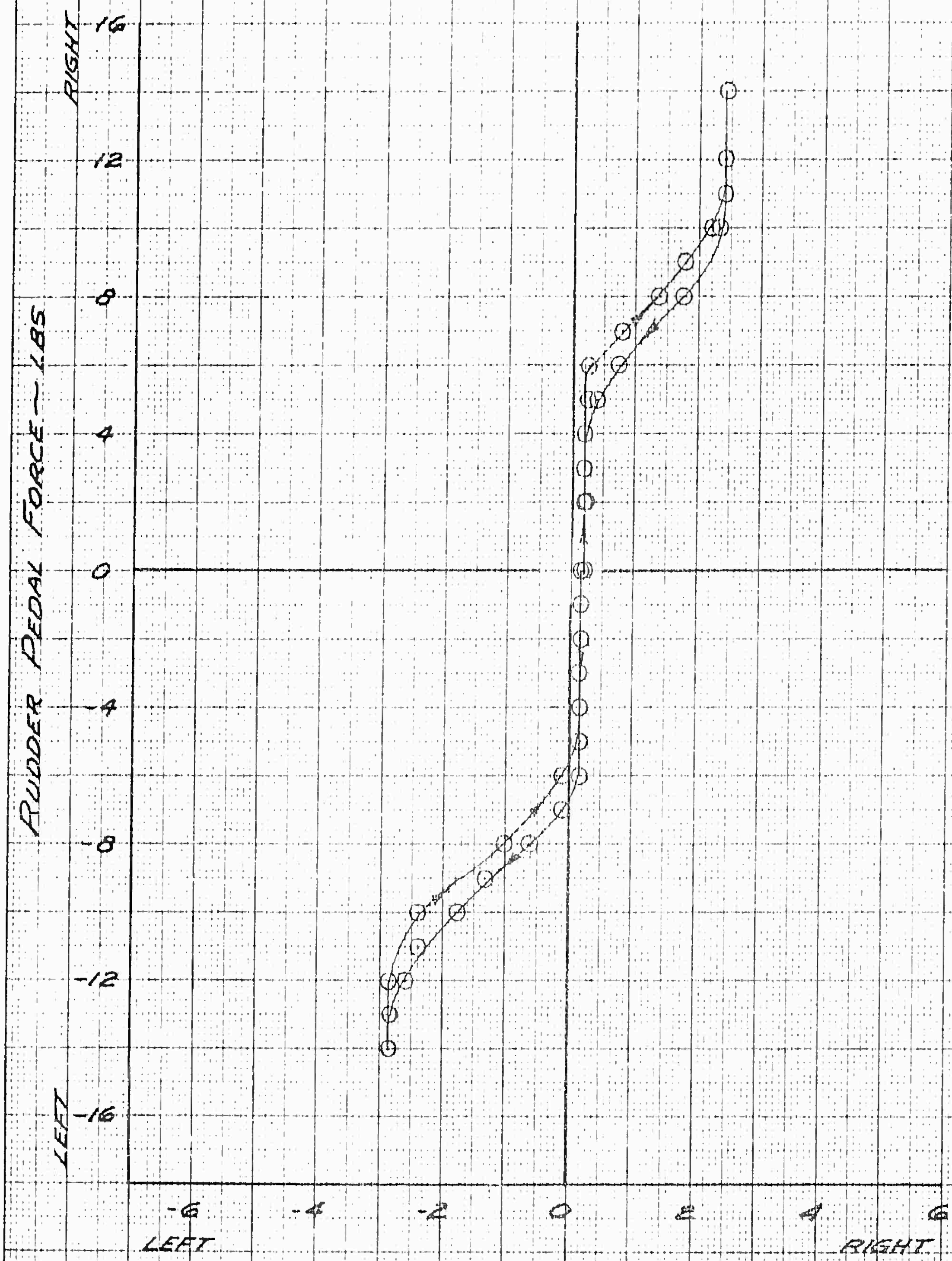
FIGURE No 95

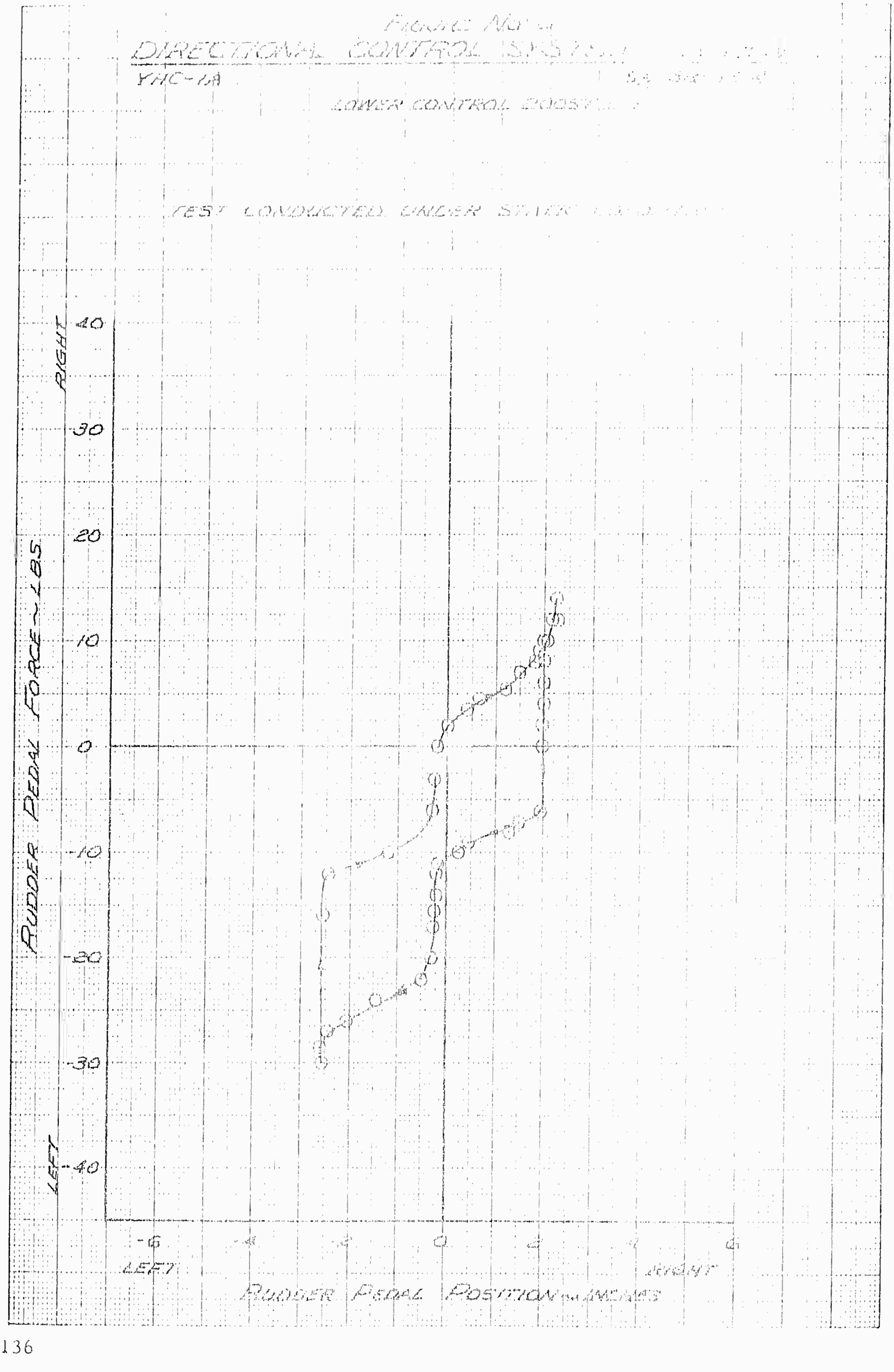
DIRECTIONAL CONTROL SYSTEM FRICTION

YHC-1A S/N 58-5514

CONTROL BOOST ON

TEST CONDUCTED UNDER STATIC CONDITIONS





DIRECTIONAL CONTROL SYSTEM
YHC-1A
TEST CONDUCTED UNDER STATIC
RIGHT
40
30
20
10
0
-10
-20
-30
-40
LEFT
RUDDER PEDAL FORCE ~ LBS
-6
-4
-2
0
2
4
6
LEFT
RIGHT
RUDDER PEDAL POSITION ~ INCHES

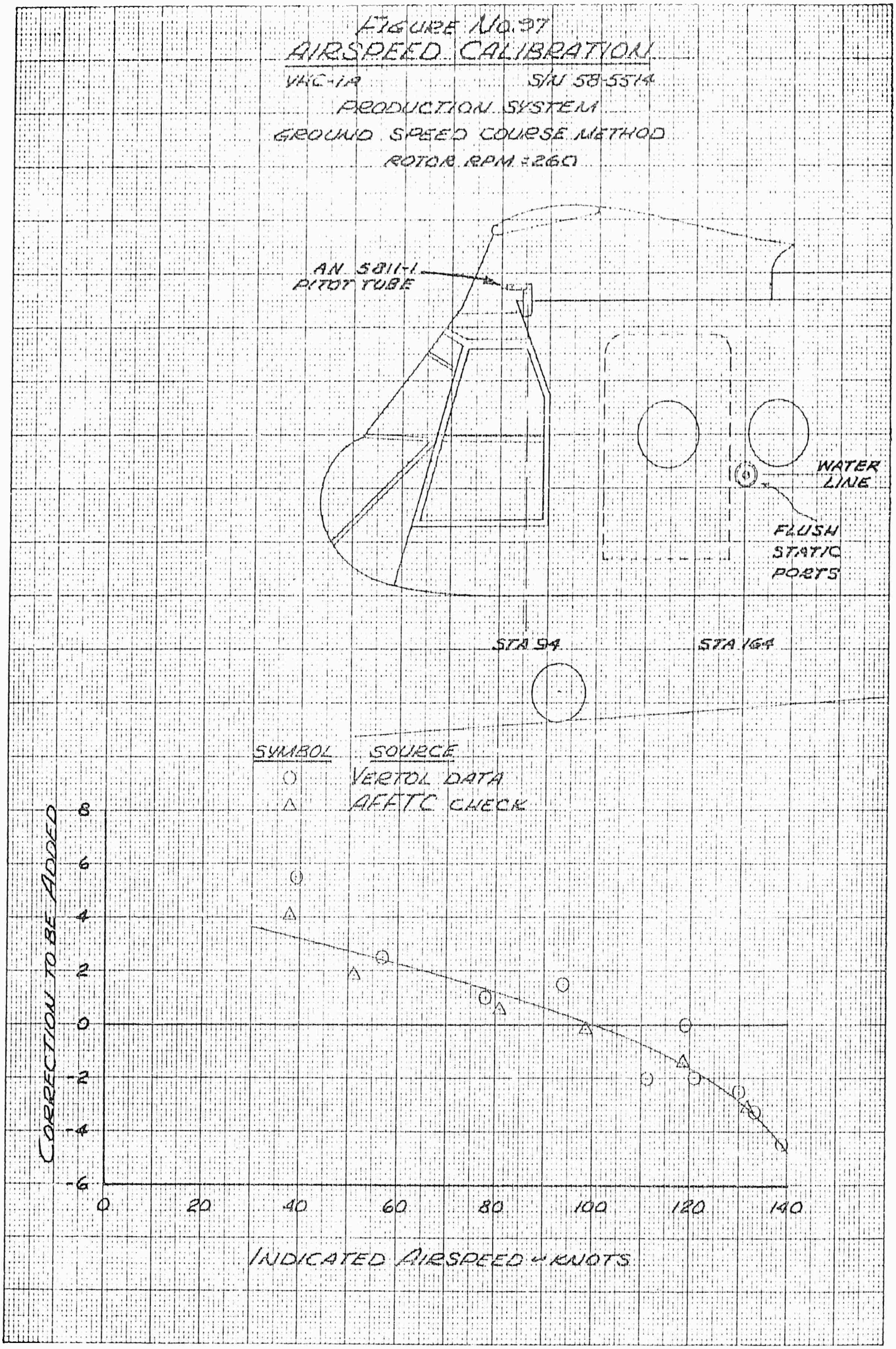


FIGURE NO. 37
AIRSPEED CALIBRATION
YHC-1A
S/N 58-5514
PRODUCTION SYSTEM
GROUND SPEED COURSE METHOD
ROTOR RPM = 260
AN 5811-1
PITOT TUBE
WATER
LINE
FLUSH
STATIC
PORTS
STA 94
STA 164
SYMBOL
SOURCE
VERTOL DATA
AFFTC CHECK
CORRECTION TO BE ADDED
8
6
4
2
0
-2
-4
-6
0
20
40
60
80
100
120
140
INDICATED AIRSPEED ~ KNOTS

APPENDIX II

general aircraft information

FLIGHT CONTROL SYSTEM

Longitudinal:

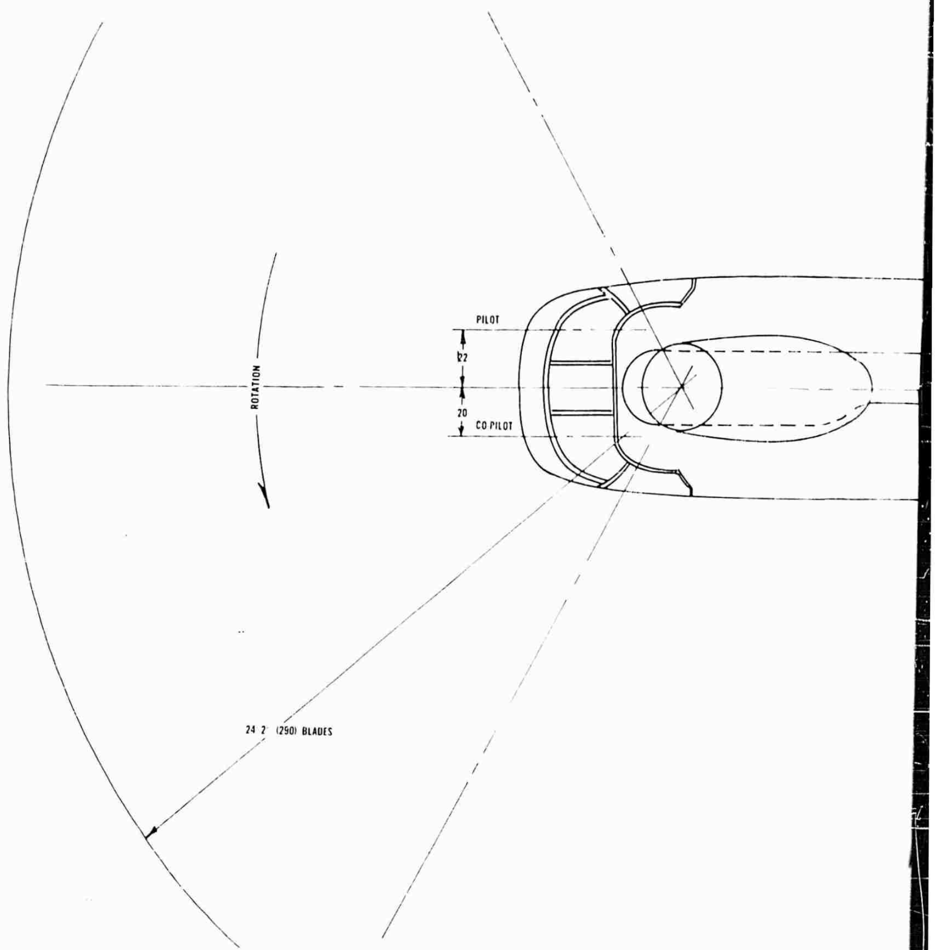
Pitch control of the YHC-1A helicopter is by means of longitudinal stick motion. Longitudinal stick travel of ± 6 inches results in ± 3 degrees of collective pitch applied differentially to the forward and aft rotors. The sense is such that a forward stick motion will result in a decrease in collective pitch at the front rotor and an equal increase at the rear.

Power boost is provided in two stages. A hydraulic boost actuator is provided in the control system at the point where longitudinal stick output remains separate from any other motions. The second boost stage is in the individual rotor controls where the inputs are "mixed", i.e., they contain not only the differential collective pitch motions from the longitudinal stick, but also collective pitch motions from the collective stick. The upper hydraulic actuators are irreversible with respect to rotor forces, whether boost is on or not. A completely separate hydraulic system is provided for each boost and are referred to as the lower hydraulic system for the lower boost, and the upper hydraulic system for the upper boost.

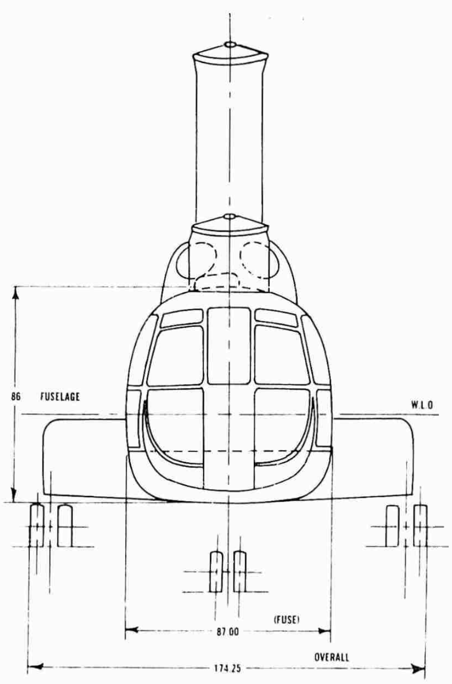
A rate damper system is incorporated in the longitudinal control axis. A signal is taken from the pilot's (instrument) vertical gyro, and by means of electrical networks, pitch rate is derived, given a time lag, and fed into a differential hydraulic actuator on the input side of the main hydraulic boost. Control motions which oppose aircraft pitch velocities are therefore fed to the rotor controls without causing motion of the pilot's stick. This differential actuator is limited in authority to 25 percent of full control travel.

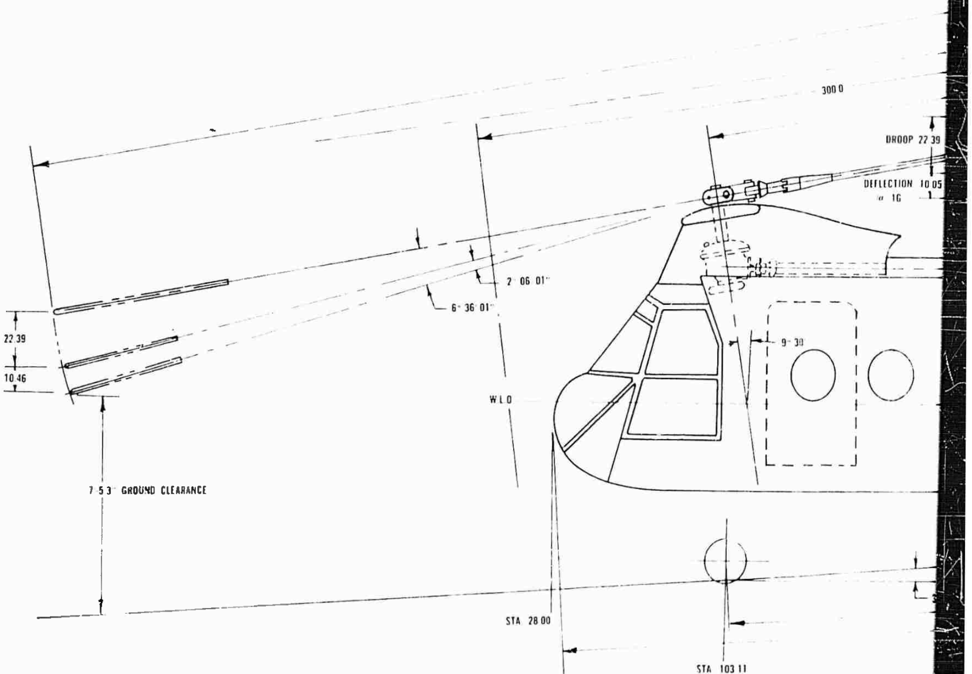
Over-travel of the pilot stick is provided so that with the actuator at either extreme, complete control remains. A completely dual system is achieved making use of signals from the copilot's vertical gyro fed through duplicated electrical networks and fed into hydraulic extensible links above the primary boost. The differential actuators which feed into the lower boost hydraulic system are powered by the lower hydraulic system and the extensible links which provide inputs to the upper boost are powered by the upper hydraulic system, thereby achieving the reliability of a completely duplicated system.

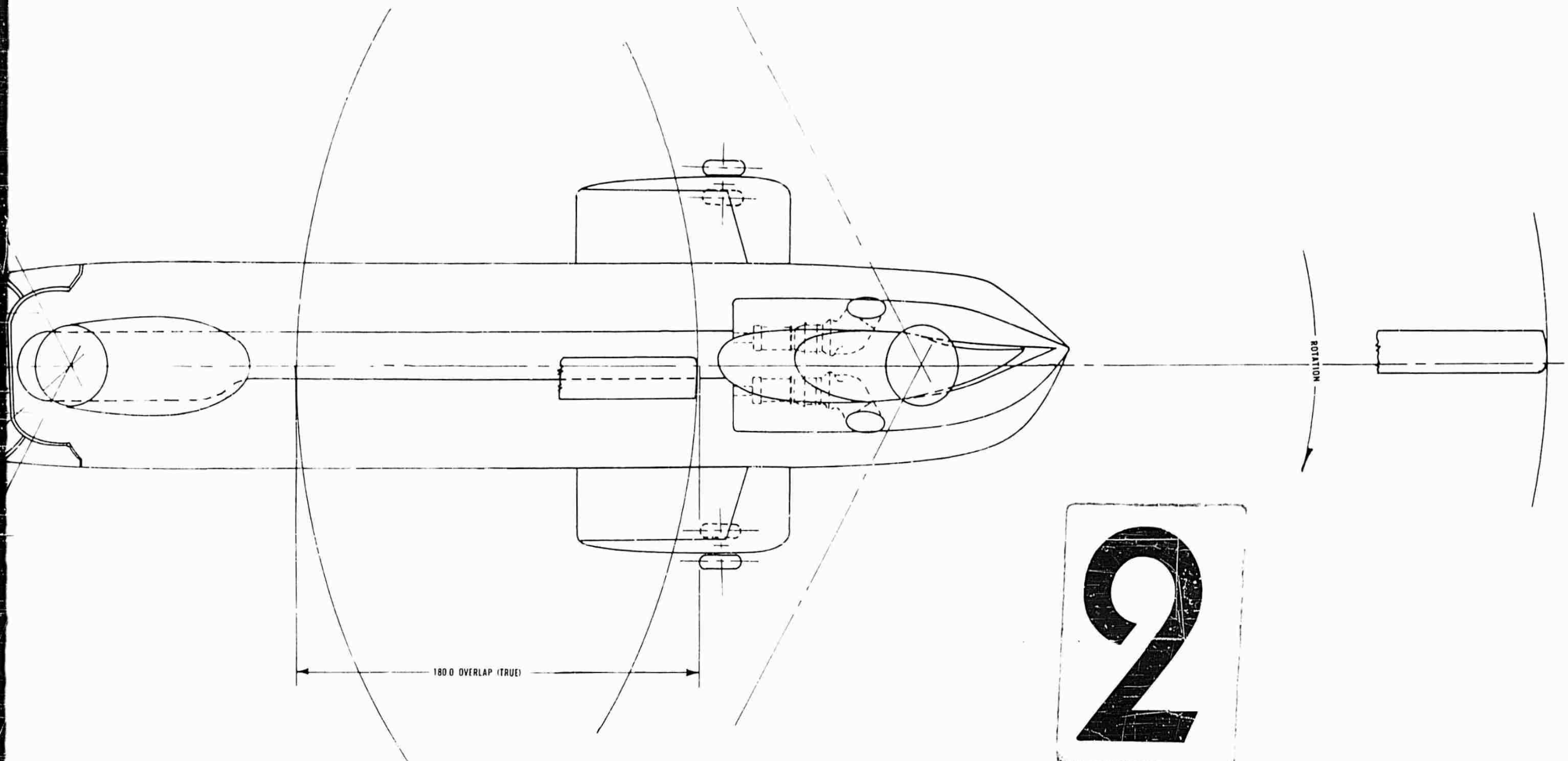
Swept Disc Area	3,670 Sq. Ft
Projected Disc Area	3,310 Sq. Ft
Percent Overland	31°
Blade Twist	8° 3'
Airfoil Section	NACA 012 Modified
Transmission Ratio	73,772 1.0
Rotor Solidity	.0594
Number of Blades	3
Normal Gross Weight	15,550 Lbs.
Weight Empty	9075 Lbs.
Most Fwd Cg	28.0 Fwd C L
Most Aft Cg	10.0 Aft of C L
Rotor Disc Loading	4.24 Lbs./Sq. Ft
Blade Pitch Range	– 17.5° to 35.5°
Coning Stop Angle	+ 30°
Collective Pitch	1° to 17°
Droop Stop Angle	– 4.5 Max.



1

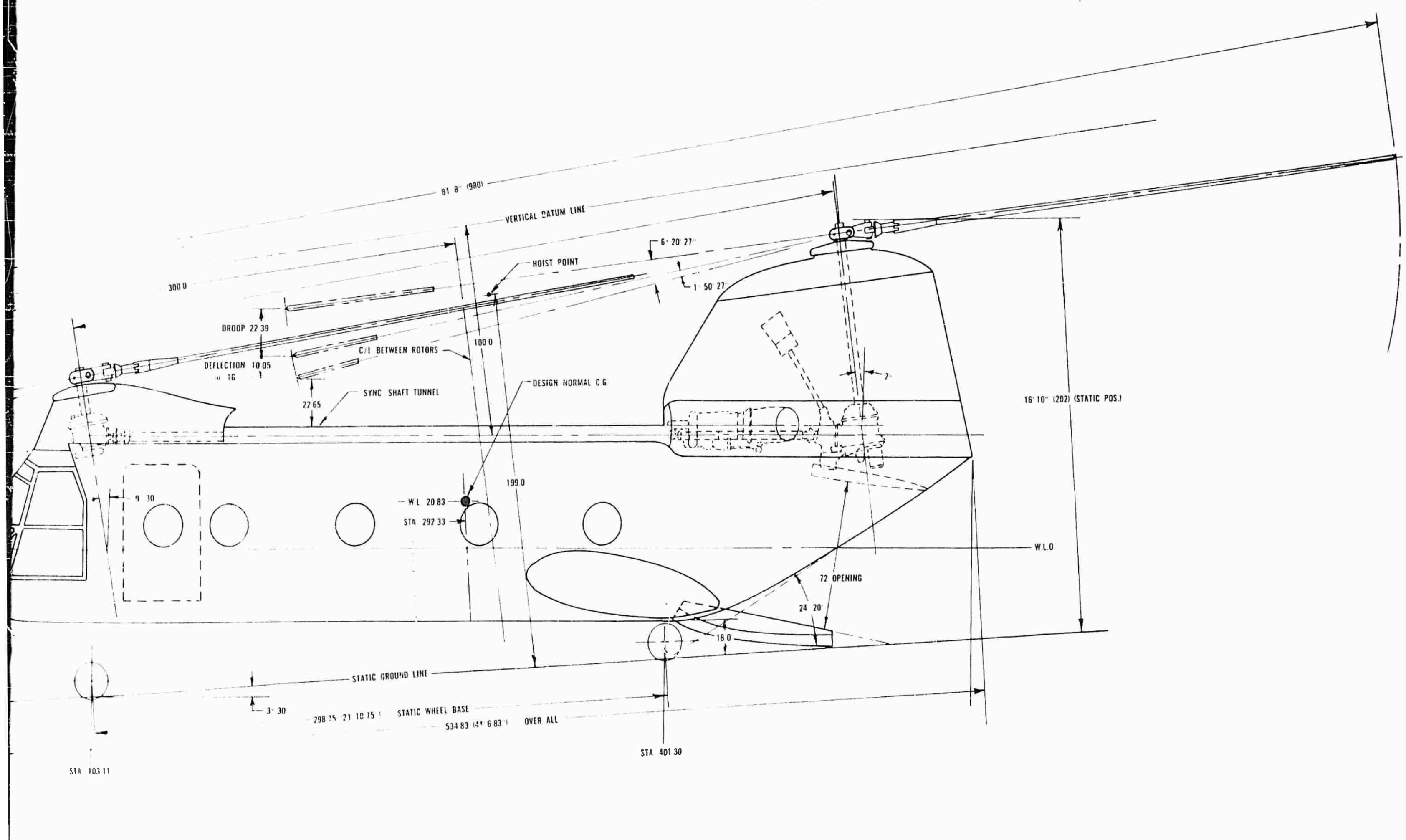






ROTATION
180.0 OVERLAP (TRUE)

2



81' 8" (980)
VERTICAL DATUM LINE
6° 20' 27"
HOIST POINT
300.0
1° 50' 27"
DROOP 22.39
100.0
C/L BETWEEN ROTORS
DEFLECTION 10.05
@ 1G
DESIGN NORMAL C.G
SYNC SHAFT TUNNEL
22.65
7°
16' 10" (202) (STATIC POS.)
9° 30
199.0
W.L. 20.83
STA 292.33
W.L.O
72 OPENING
24° 20'
18.0
STATIC GROUND LINE
3° 30
298.75 (21' 10.75") STATIC WHEEL BASE
534.83 (44' 6.83") OVER ALL
STA 401.30
STA 103.11

A four position switch allows selecting both upper and lower rate systems operating together (normal operation) either the upper or lower operating alone, (pitch, roll and yaw) function at once.

Longitudinal cyclic control from 0.5 degrees aft to a maximum of 7.5 degrees forward is provided for the rear rotor. This is controlled automatically by a total pressure transducer which motivates a hydraulic trim actuator at the rear rotor. From 0 to 60 knots CAS, cyclic is in the most aft position and from 60 knots it increases approximately linearly to full forward at 135 knots. An internal mechanical spring is built into the longitudinal cyclic actuator so that in the event of a hydraulic failure the actuator will automatically go to the full aft position at a rate determined by a hydraulic orifice.

A "beep" trim system which allows the pilot to put differential collective pitch into the rotors by means of a linear electric actuator is mounted just above the longitudinal extensible link. Since this is actually the same control as longitudinal stick, use of this trim merely repositions the stick. The equivalent of ± 2 inches of stick is available so that the stick may be positioned as the pilot desires depending on aircraft gross weight, c g or flight condition.

Mag-brakes are provided in longitudinal, lateral and directional axes which allow the pilot to trim control forces to zero at any position by pressing the mag-brake button.

The collective pitch stick has a 12 inch travel and puts 1 to 17 degrees pitch, measured at the 3/4 radius station, simultaneously into all blades of both rotors. The boost system is identical to that of the longitudinal system.

Lateral:

Lateral motion of the stick of ± 3.6 inches results in ± 7.27 degrees of lateral cyclic pitch to the forward rotor and ± 4.73 degrees lateral stick to the aft rotor. The sense is such that motion of the stick to the right tilts both rotors to the right. This provides essentially lateral control to the aircraft; the slightly greater control on the forward rotor than the aft is incorporated so as to improve the aircrafts ability to make "co-ordinated stick turns" in forward flight.

The boost system for the lateral control axis is essentially identical to that for the longitudinal control axis except that the second upper power boost actuators receive a mixture of lateral and directional control.

The roll rate damping system is the same as the pitch rate damping system except that the derived rate signal is not lagged in the lateral control axis.

Directional:

Directional control is achieved through lateral cyclic pitch applied differentially at the two rotors. Rudder pedal displacements of ±2.3 inches result in ±7.13 degrees of lateral cyclic at the rotor heads. The sense is such that a forward displacement of the right rudder pedal tilts the front rotor to the right and the rear rotor to the left. The rudder pedals have a "unitary adjustment" which allows them to be moved together ±3 inches in the fore and aft directions.

The power boost system in the directional axis is the same as in lateral axis.

An electrically driven rate gyro is provided, from which a stabilizing signal is obtained which is amplified and fed into the lower directional differential actuator. This lower actuator as well as the upper extensible link has an authority of 33 percent of the total control travel. As in the case of the pitch roll axis complete over-travel is provided in the cockpit so that no control is lost even in the event of a hard-over failure.

This system is completely dualized and the output of the duplicate fed into the extensible links. In order that this system not oppose the pilot during long, steady turns, the yaw rate signal is electrically "washed out" so that only rather rapid changes in yaw rate are damped. In addition, two static pressure ports located under the windscreen on either side of the axis of symmetry of the aircraft are used to provide sideslip stability in order to improve the co-ordinated stick-turn characteristics of the aircraft. Pressures sensed by these ports act on either side of the same diaphragm and an electrical transducer provides a signal proportional to the differential pressure and, therefore, to the aircraft sideslip. This signal along with the yaw rate signal is fed into the lower directional differential actuator. A completely duplicated sideslip sensing system is provided and fed into the extensible links. Finally, in order that entrance into and out of stick-turns be made rapidly, a cross signal from roll attitude into yaw is provided so that as soon as a roll angle is established, directional control in the co-ordinated stick-turn sense, is provided to the rotors. This signal is, of course, also duplicated since there are vertical gyros for both pilot and copilot.

POWER PLANT

Power for the rotors is supplied by two General Electric YT-58-6 rear drive gas turbine engines, each capable of delivering 1050 SHP (military rated power) at sea level, or 900 SHP (normal rated power) at sea level. The YT-58-6 engine features a free power turbine, i.e., there is no mechanical connection between the power turbine and the gas generator. Accordingly, the power turbine is connected by a short shaft directly to the combining gear box portion of the aft transmission and thereby power is transmitted to the rotor drive system.

The engines are housed side by side in separate stainless steel compartments in the base of the aft pylon above the rear loading ramp. Combustion air for each engine is provided by individual inlet ducts in the leading edge of the pylon. Each engine is supported at two points with quick disconnect mounts. The forward mount is on the top of the engine and takes vertical and side loads. The aft mount takes loads in all directions and is a torque tube with one end bolted to the aft face of the engine and the other end fastened to a quick disconnect on the combining gear box support. The exhaust from each engine is directed aft and outboard through an opening in the pylon skin. The cooling air for each engine compartment is ducted in through a scoop in the forward outboard portion of each compartment. This air flows through the engine compartment and is discharged overboard through ejector action of the engine exhaust. A continuous type fire detection system is installed in each compartment.

Engine oil is supplied from a twin hopper tank mounted in the leading edge portion of the aft pylon. The oil is cooled through separate oil coolers which are packaged with the transmission oil coolers and uses the same blower system. The fuel is supplied from two fuel tanks mounted at the leading edge assemblies of the right hand and left hand stub wings. These assemblies are designed as quickly removable units being attached to the fuselage and the landing gear support structure by four bolts. Each tank consists of a sheet metal structural shell which contains a bladder type cell of crash resistant material. The volume of each tank is 180 gallons. The left cell supplies the left hand engine, the right cell supplies the right hand engine and a pilot controllable fuel cross-feed system allows both tanks to feed one engine. Each tank is provided with an integrally mounted, electrically powered, fuel boost pump capable of delivering required fuel flow to the engine driven fuel pump inlet throughout the operating range of the engine. An airframe mounted, 40 micron fuel filter with impending by-pass indication for the pilot is also provided for each tank system.

Engine access is provided through hinged stainless steel, soundproofed doors which form the bottom of each engine compartment and swing down and outboard when opened. Engine removal is accomplished by a standard bomb hoist attached in the leading edge area of the pylon and so positioned as to permit the engine to be lowered directly onto the ramp.

Engine Control System:

The engine control system is electrically operated and consists of a semi-automatic signal integral engine control system located on the accessory section of each engine; two engine condition levers located on the console in the pilots' compartment; two engine speed selector (beep trim) switches located on each collective pitch lever and two airframe mounted electric actuators which transmit cockpit changes to the engine control.

The engine's control system automatically maintains and governs engine output speed regardless of helicopter power demand by regulating fuel flow to increase or decrease gas generator speed as required and automatically regulates variable stator vane position during starting.

The engine condition levers are electrically connected to actuators, located in the engine compartment, which are mechanically connected to the engine's control systems. Each lever in the cockpit is provided with five labeled positions: "STOP", "START", "G.I." (Ground Idle), "F.I." (Flight Idle), and "FLY". Each position corresponds to the proper position on the engine control levers. When the engine condition levers are moved to the "FLY"position, prior to tkae-off, the pilot is allowed to select desired rotor rpm with the engine speed selector (beep trim) switches, and the engine's control system accordingly meters fuel to the gas generator so that the power turbine delivers power as required to the rotors at constant speed.

The engine speed selector (beep trim) switches are electrically interconnected with the engine condition levers and permit the pilot to synchronize engine output power and to adjust rotor rpm, as desired, when the engine condition levers are in the "FLY" position. The switch labeled "BOTH ENG" simultaneously controls bothengines, and is used to change rotor rpm. The switch labeled No. 2 controls No 2 engine only, and is used to match output power of the No. 1 engine.

Drive System:

The drive system consists of the aft transmission assembly, forward transmission assembly, interconnecting shafting, rear rotor shaft and the lubrication system. The aft transmission assembly contains a combining gear box sub-assembly which receives the power from each engine at 19,500 rpm and reduces this speed to 2500 rpm for the interconnecting shafting. Within the combining box, a sprag over running clutch is provided for each side to prevent the extra drag of the power trubine in the event of an engine failure. This combining box has been designed and bench tested for single engine operation at 1250 SHP and for continuous operation at a total of 2100 SHP. The other sub-assembly of the aft transmission contains the spiral bevel gears, planetary gear system and accessory section. External lubrication lines on the aft transmission have been virtually eliminated by the use of integral "cast in place" oil passages in the case castings. The forward transmission is a modification of the H-21C rotor transmission, essentially the same as that done for the YH-21D program. The design of each rotor transmission is such as to permit it to take 60 percent of the combined transmission rating of 1700 SHP.

The speed of the interconnecting shafting is reduced to rotor rpm through the spiral bevel gears and the planet gears of the forward and aft transmissions. The forward rotor shaft is an integral part of the forward transmission, whereas the aft rotor shaft is splined to the aft transmission and a thrust bearing is mounted on the shaft to take the thrust loads out directly at the top of the aft pylon. The shafting interconnecting the two transmissions consists of five shock mounted sections connected by flexible Thomas couplings. Four of these shaft assemblies are interchangeable. The engine drive shaft, operating at 19,500 rpm, is attached to the engine supplied "polygon coupling." The other end of the shaft is attached to the combining box through a Thomas coupling.

The accessories on the forward transmission include the two hydraulic pumps for the control system, a lubrication pump and a rotor brake. Similarly, the aft transmission drives the oil cooler blower, a lubrication pump, a tachometer generator, two 20 KVA alternators and has provisions for a square pad. The lubrication system is supplied with oil from the integral pumps of each transmission and the oil is cooled through the cooler package mounted in the leading edge of the aft pylon.

ROTOR SYSTEM

The rotor configuration is the overlapped tandem type consisting of two 48' 4" diameter 3-bladed, fully articulated rotors, synchronized by positive gearing and interconnecting drive shafts. Rotor blade design is the Vertol seamless steel D-spar metal blade with turned root end, 18 inch blade chord, -8.3 degrees blade twist and a modified NACA 0012 airfoil section. The airfoil aft of the D-spar is made up of aluminum boxes individually bonded to the spar and with a 1.5 inch extension along the entire trailing edge in the form of a continuous tab. The forward blades are interchangeable with each other and the aft blades are interchangeable with each other. The hub assembly is essentially the same as that of the H-21C. The horizontal pin is mounted in the hub and is connected to the vertical pin by an extension link. Outboard of the vertical pin is the pitch bearing housing, tension-torsion straps, blade attachment, pitch arm and lag damper. The rotor controls are also essentially the same as the H-21C and consist of the swash plate assembly, drive arm, gimbal ring and pitch links.

ELECTRICAL SYSTEM

The main electrical power for the helicopter is generated by two, 20 KVA ac alternators which give 208V, 3-phase, 400 cycle current. These alternators are driven from the accessory section of the aft transmission and are non-paralleled. The a.c. electrical distribution center is located in the trailing edge structural cavity of the right hand stub wing and consists of the generator control panels, automatic changeover control, relays and an instrument inverter. Correspondingly, the d.c. electrical distribution center is located in the left hand stub wing and consists of a 24V, 34 ampere hour nickle cadmium battery, transformer-rectifiers, a d.c. distribution panel and two external power receptacles, one for a.c. and one for d.c. The wiring to the forward end of the helicopter is routed in a fiberglas conduit which is just above the covering on the left hand side of the cabin.

The console is located between the pilots and contains the radio control panels, engine condition levers and the miscellaneous required switch panels. Circuit breakers are located in the side panels of the console and in a box at the aft end of the cabin. The instrument panel contains basic flight instruments for each pilot with engine and transmission instruments mounted in the center section and shared by the pilots.

EQUIPMENT

The cabin and the cockpit of the helicopter are fitted with insulating blankets of both thermal and acoustical capability. A 200,000 BTU heater and blower system is provided to supply heat and fresh air to the cockpit and to the cabin. The heater is located in the forward end of the cabin under the floor of the passageway to the cockpit. Defrosting and defogging air can be directed onto the windshield at the pilot's option and valves are provided in the ducts which permit the pilot to direct all air to the cockpit and/or to the cabin.

The windshields for the pilot and the copilot are laminated plate glass with electric windshield wipers for each pilot. The plexiglas panels adjacent to each pilot are jettisonable for emergency escape and present an opening from floor level to the overhead. The pilots' seats are adjustable fore and aft and vertically and are equipped with seat belts, safety harnesses and provisions for parachutes or cushions. In the passageway between the cabin and the cockpit is a stowable jump seat and safety belt for the troop commander or crew chief.

The pitot static system consists of a total head mast mounted over the cockpit on the left hand side and a flush static port mounted on each side of the fuselage near the maximum half-breadth and just aft of the main cabin door. The external lights on the helicopter include two rotating anti-collision lights, one on the bottom of the fuselage and one on the top of the rear pylon, and running lights on each side and at the tail. Internal lights consist of cabin dome lights, cockpit utility lights, instrument lights and console panel edge lighting.

The electronic equipment installed in this helicopter includes the ARC-44, ARC-55, ARN-59, ARN-32, ARA-31 and provisions for APN-117, ARN-30A and APX-44. Also included is the interphone system and cabin loudspeakers. The antennae associated with the above equipment is also installed or provided for in the helicopter. The electronic equipment is installed in two compartments, one in the nose of the helicopter and one in the forward end of the cabin on the left hand side just behind the co-pilot's bulkhead.

Two retractable rear view mirrors are installed at floor level just outboard and forward of each pilot. Each mirror is controllable from a pilot's control (C.P.) stick and enables the pilot to view rearward and under the helicopter to the area occupied by external cargo carried on the cargo hook. The cabin is outfitted for carrying either cargo or personnel. The flooring has 200 pound/sq.in. capacity and is of honeycomb construction with a metal, non-skid surface which also contains flush fittings for a combination of cargo tiedown, troop seat retention and litter support retention. Twenty two troop seats with individual safety belts are provided for installation in helicopter. There is also space provision for four additional troop seats aft of the twenty-two seats. The troop seats may be folded up against the side wall or removed, thus permitting the installation of the litter supports which will accommodate 15 litters. With the litters installed, there are seats provided for two medical attendants and the aisle between the two banks of litters permits easy servicing of the patients.

In the center of the cabin just under the helicopter c g and located in the floor is a hatch opening which is used during external cargo load missions and during rescue hoist missions. This opening is closed at the skin line by two doors which are operated manually from inside or outside the helicopter. The opening is closed over at the floor line by hinged floor panels which carry the design floor load when closed. For external cargo use, a beam to which is bolted a conventional 5000 pound capacity cargo hook, is mounted fore and aft across the opening in trunnion type sockets. In this manner, the hook hangs down at about keel level and is free to rotate in the trunnions as the load swings. As the helicopter hovers over a load the crewman reaches down through the opening with a "boat hook" and engages the cargo ring into the cargo hook. The "boat hook" is electrically bonded to the helicopter, thus preventing static discharge through the crewman's body. A load may be released by one of several ways, normal release from pilot's control stick, touchdown release, emergency release by pilot or crewman and manual release by crewman. When not in use, the hook-beam assembly is stowed by rotating it into a horizontal position or by quick removal.

A utility load handling winch is located at the forward end of the cabin in the overhead. The cable from the winch runs aft in the crown of the cabin to a quickly detachable pulley assembly located directly over the hatch. This arrangement permits rescue operation through the hatch opening. For ground loading operations, the detachable pulley is hooked onto a load which is on the ground aft of the ramp and the end of the cable is carried back into the cabin and secured at the forward bulkhead. The rating of the winch for rescue work is 600 pounds at 100 ft/min. or 1200 pounds at 50 ft./min. For the cargo loading, the cable arrangement reduces the speeds in half and permits a pull of 1200 pounds at 50 ft./min. or 2400 pounds at 25 ft./min.

LANDING GEAR

The tricycle landing gear configuration consists of two main gear assemblies of the fixed vertical type and a nose landing gear assembly of the castering type. Each gear has a conventional fixed oleo, air-oil shock absorbing strut. Each main gear is provided with 18 x 5.5 wheels and brakes. The tires are 18 x 5.5 Type VII, 8 ply rating, and are tubeless with side wall inflation. The nose gear is provided with the same basic wheels and tires as for each main gear except that the wheels are co-rotating so as to eliminate the need for a shimmy damper. The nose gear is self-centering and also has a cam detent device which locks the gear in the trailing aft position. The brakes on the main wheels are operated by the pedals on the pilot's rudder pedals.

WEIGHT AND BALANCE

The test aircraft was weighed prior to initiation of the test program. The empty weight of the aircraft was 12,530 pounds including full oil, no trapped fuel, instrumentation, and an empty ballast box. The center of gravity for this condition was 5.7 inches forward of the centerline between the rotors. The instrumentation installed in the test aircraft weighs approximately 3137 pounds and results in a lateral c g offset of .53 inches to the right of the centerline.

The basic gross weight of a standard YHC-1A helicopter is 8946 pounds with a fuel capacity of 345 gallons (2240 pounds).

TEST INSTRUMENTATION

The test instrumentation utilized during this program was the existing instrumentation as installed in the aircraft by the Vertol Division of the Boeing Airplane Company. Calibration spot checks were made prior to the initiating of flying. A large quantity of instrumentation was installed in the aircraft which was not required for this flight evaluation and only those parameters utilized are listed in this section.

A photo recorder and four oscillographs were installed in the aircraft and the parameters recorded are as follows:

Photo recorder
- airspeed
- altitude
- rotor speed
- gas producer speed (both engines)
- fuel flow (both engines)
- turbine inlet temperature (both engines)
- fuel flow
- turbine inlet temperature (both engines)
- fuel used
- free air temperature
- angle of sideslip
- control positions

Oscillograph
- rotor torque (fwd and aft)
- free air temperature
- gas producer speed (both engines)
- turbine inlet temperature (both engines)
- exhaust gas temperature (both engines)

sideslip angle
angular acceleration in roll, pitch, and yaw
angles of bank and pitch
rates of roll, pitch, and yaw
control position
c g linear acceleration (normal)
speed trim position
differential collective pitch trim
vibration accelerometers at cockpit floor
vibration accelerometers at c g
compresser inlet total temperature

FLIGHT LIMITATIONS

The following is a list of flight limitations as imposed by the contractor and the Navy during this evaluation.

Maximum power on rotor speed	270 rpm
Maximum power off rotor speed	320 rpm
Minimum rotor speed	240 rpm
Maximum take-off to gross weight	15,800 pounds
Maximum gross weight for hovering	16,700 pounds
Maximum load factor	1.5 g
Minimum load factor	1.0 g
Forward c g limit at 15,550 pounds	24 inches fwd*
Aft c g limit at 15,550 pounds	5.5 inches aft

*Measured from the centerline between the two rotors.

In addition the following airspeed restrictions were placed on the test helicopters.

Rotor rpm	Knots CAS
240	80
250	95
260	140
270	140

In rearward and sideward flight the helicopter was restricted to 20 knots.

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